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AD859805

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USAAVLABS TECHNICAL REPORT 69-14

A STUDY OF AERODYNAMIC PERFORMANCE EVALUATION AND COMPARISON TECHNIQUES FOR V/STOL AIRCRAFT

By

Donald W. Boatwright

July 1969

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-376(T)
THE DEPARTMENT OF AEROPHYSICS AND AEROSPACE ENGINEERING
MISSISSIPPI STATE UNIVERSITY
STATE COLLEGE, MISSISSIPPI

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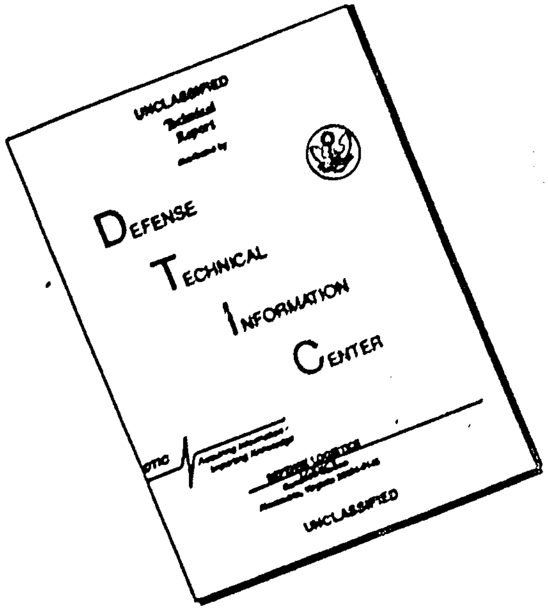

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This report was prepared by the Department of Aerophysics and Aerospace Engineering, Mississippi State University, under the terms of Contract DA 44-177-AMC-376(T).

The report has been reviewed by the U. S. Army Aviation Materiel Laboratories and is published for the dissemination of information and the stimulation of research.

Task 1F162204A14234
Contract DA 44-177-AMC-376(T)
USAAVLABS Technical Report 69-14
July 1969

A STUDY OF AERODYNAMIC PERFORMANCE EVALUATION AND COMPARISON TECHNIQUES FOR V/STOL AIRCRAFT

Aerophysics and Aerospace Research Report No. 77

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ABSTRACT

This report presents the results of an investigation of performance evaluation and comparison methods as applied to V/STOL aircraft. Attention is given to a number of aircraft having different sizes, gross weights, geometric configurations, and propulsion systems. Particular regard is given to the use of thermal fuel energy as a common basis for evaluation and comparison of V/STOL performance. The performance capabilities of typical V/STOL aircraft are presented and compared using both dimensional and nondimensional parameters containing fuel flow rate as a variable. In addition, three aircraft of different configurations are analyzed with regard to the effects of altitude, gross weight, and payload-to-fuel load ratio on the performance capability of each aircraft as indicated by both new and conventionally used methods. The total energy concept is discussed with regard to the optimization of climb schedules, and consideration is given to the limitations which apply to the use of nondimensional parameters which are used to describe the flow regimes of V/STOL aircraft.

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a	speed of sound - ft/sec
C_D	total aircraft drag coefficient - dimensionless
C_L	total aircraft lift coefficient - dimensionless
D	total aircraft drag - lb
E	endurance - hr
g	gravitational acceleration - ft/sec^2
h	true altitude - ft
h_e	energy height - ft
$(HP)_T$	thermal power - hp
k	constant - nautical miles
ℓ	length - ft
L	total aircraft lift - lb
MGW	maximum gross weight - lb
MPL	maximum payload - lb
MPT	maximum performance takeoff - dimensionless
MRP	maximum rated power - hp
NRP	normal rated power - hp
Q	rate of fuel consumption - lb/hr unless otherwise specified
q	fuel heat value - Btu/lb
R	range - nautical miles
R/C	rate of climb - ft/min unless otherwise specified
S	area - ft^2
SHP	shaft horsepower - hp

T	thrust - lb
THP	thrust horsepower - hp
t	time - sec
V	true airspeed - kn unless otherwise specified
W	weight - lb
η	efficiency - dimensionless
μ	dynamic viscosity - lb-sec/ft^2
ρ	density - slugs/ft^3

Subscripts

e	equivalent
f	fuel
g	gross weight condition
min	minimum
max	maximum
0	total condition
p	fuel plus payload
pr	propulsive
T	thermal
u	payload
o	zero acceleration along flight path

INTRODUCTION

Since the helicopter began to be widely employed in military aviation almost two decades ago, a number of new types of VTOL and STOL vehicles have emerged and have presently reached a stage of development which presents a challenge to the helicopter and its role in military operations. While the helicopter will undoubtedly continue to play an important part in both commercial and military aviation in future years, V/STOL technology is making rapid strides. New types of VTOL and STOL vehicles now appear to show promise of becoming as important to aviation as the helicopter has been in past years. Already a dozen or more new V/STOL-type aircraft of widely varying configuration and capability are in the flight-test stage of development for use as military aircraft in this country and abroad.

Because V/STOL vehicles derive their high-lift capability in a variety of ways and because of the many different configurations that may be expected to exist in the future, it should not be unreasonable to expect that current problems of evaluating and comparing the performance capabilities of these aircraft will continue to exist unless new evaluation techniques and parameters are devised. These new techniques must account for the integrated role of the power plant system and aerodynamic configuration and must be applicable to aircraft for a variety of flight regimes and mission, range, and payload requirements.

In view of the rapid developments being made in V/STOL technology and the promising role of V/STOL aircraft in military aviation, the present study was initiated to investigate current performance evaluation methods used in the V/STOL field by manufacturers and testing agencies. The objectives were to determine the limitations, assumptions, and problems

which exist in the evaluation and comparison of the aerodynamic performance of V/STOL vehicles. It was also the purpose of this study to examine new methods that would allow comparison of the aerodynamic performance of various types of V/STOL aircraft and which would be applicable to all aircraft regardless of configuration or propulsion system.

METHODS OF AIRCRAFT PERFORMANCE EVALUATION

This investigation was devoted to a study of various methods of evaluating aircraft performance which would permit meaningful comparisons of the performance capabilities of V/STOL aircraft. A fixed-wing aircraft (YAC-1DH "Caribou") was included in the analysis to allow comparisons of fixed-wing performance with V/STOL performance (Reference 16).

The methods and techniques of evaluating V/STOL aircraft performance that were considered in this study were as follows:

TOTAL ENERGY METHOD

The "total energy method" discussed here refers to the technique used to optimize aircraft performance between given initial and final conditions. This technique is based on the total mechanical energy possessed by an aircraft; i.e., the sum of the potential and kinetic energies of the aircraft at a given time. The method as presently developed (Reference 4) may be used to establish the velocity-height schedule that is required for an aircraft in passing from initial to final conditions in a minimum amount of time. It is also shown in Reference 4 that use of the total energy concept to determine maximum range provides only small deviations from the velocity-height schedules for minimum time. As a result, use of the total energy method to optimize climb schedules for maximum range does not appear to be worthwhile in view of the small increase in total range that would be obtained from a range analysis.

Since the total mechanical energy of an aircraft is independent of configuration, the total energy method provides a means of evaluating and comparing the optimized performance of V/STOL aircraft. However, there are certain limitations to the method which prohibit its universal application to all

V/STOL aircraft. These limitations will be discussed after further considerations of the method itself.

It is shown in Reference 4 that the optimum velocity-height schedule must be such that

$$\frac{\partial(\frac{R}{C})_o}{\partial V} = \frac{V}{g} \cdot \frac{\partial(\frac{R}{C})_o}{\partial h}, \qquad (1)$$

where V is the true airspeed in feet per second, h is the true altitude in feet, and $(R/C)_o$ is the rate of climb in feet per second which would be attained at the relevant velocity and altitude with zero longitudinal acceleration. From this relationship, the velocity-height schedule may be determined which allows the aircraft to pass from initial to final conditions in the minimum time possible. $(R/C)_o$, which is a function of both altitude and velocity, is defined as

$$\left(\frac{R}{C}\right)_o = \frac{dh_e}{dt}, \qquad (2)$$

where h_e is the energy height,

$$h_e = h + \frac{V^2}{2g}, \qquad (3)$$

which is the height equivalent of the potential plus kinetic energy of the aircraft.

The analysis leading to Equation (1) is presented in detail in Reference 4 and will not be discussed here. However, it is relevant to discuss how Equation (1) may be used to obtain the optimum velocity-height schedules for V/STOL aircraft and limitations of the method.

The optimum climb schedule for a particular aircraft may be obtained by graphical means. A family of curves of $(R/C)_o$

versus true airspeed must first be obtained either from flight tests of the aircraft or from performance estimates. When these data are available, the climb schedule defined by Equation (1) is easily deduced. Flight-test data are generally obtained from sawtooth climbs or level-flight acceleration runs.

Climb schedules for the X-22A are presented in Figure 1. These schedules were calculated from estimated performance data for an aircraft weight of 15,930 pounds and for normal rated power. As illustrated in the figure, an approximation to the optimum climb schedule may be obtained by considering

$$\frac{\partial(\frac{R}{C})_o}{\partial V} = 0 \tag{4}$$

It will be observed that departure of the approximated climb schedule from the optimum schedule increases with altitude, the optimum climb velocity being higher than the approximate value. It was concluded that use of the approximate schedule would seldom be justified, since only one additional step is required to solve for the exact solution. This is accomplished by selecting a point on the energy height curves of Figure 2 and evaluating both sides of Equation (1). Several repetitions of this procedure are generally sufficient to locate the point on each energy height curve where Equation (1) is satisfied.

Figure 1 also shows a comparison of the optimum climb schedule to a climb schedule plotted from the preliminary performance data of Reference 52 at normal rated power. Although the optimum climb velocity is approximately 24.0 percent larger than that of a normal climb at 25,000 feet, differences in rate of climb are small, as shown in Figure 3.

USE OF THE TOTAL ENERGY METHOD FOR EVALUATION OF V/STOL PERFORMANCE

Several attempts were made to calculate and compare the performance of V/STOL aircraft from estimated climb data. Unfortunately, this was not possible because of a complete lack of or an insufficient amount of climbing performance data for those aircraft to which the method is most applicable. A study of the theory upon which the method is based, however, indicates that while the total energy method should serve as a useful tool in the evaluation and comparison of some types of V/STOL aircraft, only under certain conditions does the method appear to be worthwhile or of proven value.

The total energy method appears to be attractive at first glance because it is theoretically applicable to aircraft of all types, regardless of configuration. Theoretically, an aircraft flying an optimum height-velocity schedule will arrive at a given altitude and cruise velocity in minimum time. However, unless the aircraft has a climbing speed range such that the kinetic energy of the aircraft is relatively large compared to its potential energy, the time saved in reaching an assigned altitude and cruise velocity by climbing at optimum conditions will be small. As an example, an analysis of the X-22A will show that less than 1 second will be saved by climbing at optimum conditions to an altitude of 15,000 feet and then accelerating to a level speed of 185 knots. However, when the aircraft climbs at optimum speed to a velocity of 210 knots at 25,000 feet altitude, it will reach final conditions approximately 60 seconds faster than if it had climbed at near its maximum rate of climb. Since the X-22A is designed to operate at altitudes below 30,000 feet at cruise velocities of 185 knots or less, optimizing the performance of this aircraft for most normal operating conditions does not appear to be worthwhile.

It is only when the aircraft operates near its service ceiling at maximum velocity that improvements in performance gained by optimizing the climb schedule will become significant.

It was concluded that the use of total energy methods for optimizing climb performance will not ordinarily be justified for aircraft having climb speeds of less than 250 knots and which operate at altitudes of less than approximately 25,000 feet. For aircraft having high climbing speed range, such as jet-propelled fighter or interceptor-type aircraft, kinetic energy becomes very significant and the performance gained by optimization of climb schedules becomes important. Thus the choice of optimization of climb schedules will depend on the speed range and altitude of the aircraft as well as the degree of accuracy desired by the test engineer.

There are other important considerations which must be taken into account in addition to those mentioned above. One is the capability of the aircraft to maneuver such that the optimized climb schedule will be achieved in minimum time after takeoff. Likewise, the transition from the optimized schedule to level flight at the assigned cruise altitude must be achieved such that the total energy of the aircraft is as large as possible during the transition period. These requirements indicate that maneuver capability and pilot technique will play an important role in determining optimum climb performance. For aircraft climbing to relatively low altitudes, the takeoff conditions are extremely important. Also, for aircraft whose optimized climb schedules do not depart significantly from a maximum-rate-of-climb schedule, initial and final conditions will largely determine the performance gain achieved by the optimization procedure.

Another important consideration concerns the necessity to achieve flight-test data of sufficient accuracy to allow

computation of the optimum climb schedule. It may be observed in Figure 2 that the change in energy height with respect to time becomes less dependent on airspeed as altitude is increased. As a result, it becomes increasingly difficult to determine the optimum velocity at each level of energy height as altitude is increased. For this reason, flight-test data must be extremely accurate, and this accuracy will require precision flying and perhaps somewhat more sophisticated instrumentation than is normally used in evaluating climb performance.

GENERALIZED PERFORMANCE TECHNIQUES

The generalized performance method is widely used in aircraft performance analysis. The technique is used in data reduction to reduce or minimize the number of curves required to represent power and thrust for an aircraft throughout a range of altitudes. It is defined here in order to distinguish it from the other performance evaluation methods with which this report is concerned. The generalized performance method is quite useful in the analysis of performance data; however, the method itself provides no common parameter such as thermal or mechanical energy upon which performance evaluation may be based.

Most of the data presented in this report were not generalized, since it was the purpose of the author to illustrate the relative differences among performance data with respect to variables such as weight, airspeed, and altitude. However, Figure 4 is included as an illustration of the use of the generalized performance technique.

The generalized performance method is presented in many textbooks and will not be discussed in detail in this report. In general, the method is one in which thrust or power-required curves are nondimensionalized by the use of the respective values of thrust or power required at velocity for

maximum lift-to-drag ratio. For example, Figure 4 shows computed values of thrust horsepower ratio for three V/STOL aircraft operating in level flight. These values result in a single generalized power curve for the three aircraft, which is valid for all conditions of weight, configuration, and altitude. It should be noted that shaft horsepower curves for different aircraft will not reduce to a single generalized curve because of different variations of propulsive efficiency for the various aircraft.

DIMENSIONAL ANALYSIS

A number of nondimensional parameters have long been used in aerodynamic work which are useful in the organization, correlation, and interpretation of data. The parameters, which are derived using dimensional analysis methods, consist of two or more variables which describe, in some respect, a physical phenomenon. For example, Reynolds number is often used as a parameter for establishing flow criteria. Nondimensional parameters are also widely used in data presentation as a means of reducing the number of curves needed to express the relationship between a function and a number of variables. Aircraft efficiency, lift and drag coefficients, and thrust coefficients are other examples of dimensionless quantities commonly used in aerodynamic work.

In some instances, dimensionless quantities conventionally used with fixed-wing aircraft are invalid for use in the V/STOL field. Since the lift and drag of some V/STOL aircraft are not solely the result of pure aerodynamic forces, the Reynolds number becomes invalid because new variables have been introduced. In this respect, the possibility of forming new dimensionless parameters which would be applicable to all V/STOL aircraft regardless of geometric configuration or propulsion system was examined.

THE USE OF THERMAL ENERGY FOR EVALUATING AIRCRAFT PERFORMANCE

Specific range, fuel consumption, and endurance parameters have long been used as measures of the efficiency of fixed-wing aircraft. All of these parameters involve Q, the rate at which an aircraft consumes fuel. The use of Q to evaluate V/STOL aircraft is particularly applicable, due to the fact that all of these aircraft derive their V/STOL flight capabilities by an expenditure of power in some form. The fuel consumption rate Q can be expressed in terms of a fuel heat value, which in turn may be expressed as kilowatts of electrical power or in terms of other power units such as foot-pounds per hour or horsepower. Consequently, Q is a useful measure of aircraft efficiency in that it may easily be applied to aircraft having any type of power system for lift augmentation. As an example, the total power required in flight for a STOL aircraft having an electrically powered boundary layer control system could easily be found by converting the fuel flow rate and electric power into equivalent terms of horsepower. The sum of these converted values would represent the total equivalent horsepower required. The total equivalent power may then be used as a basis of comparison of the performance of the aircraft.

In considering the use of fuel flow rate as an important measure of the overall efficiency of aircraft, the following items should be noted with regard to evaluating performance capabilities of aircraft from both flight-test data and preliminary design estimates:

1. For any aircraft having lift or thrust augmentation by either utilizing or diverting the thrust of the main propulsion system, by using auxiliary power systems, or by using waste energy, the use of the fuel flow rate Q provides a means of comparing the effectiveness of the

lift or thrust augmentation on the basis of total power required. This provides a comparison of aircraft performance with respect to the aircraft's aerodynamic, thermal, and mechanical efficiencies.

2. In many instances, determining the power required during flight tests is difficult because of mechanical complexity, different types of power plants, equipment required, etc. It is easier to measure the total fuel being consumed by the aircraft regardless of the number or type of engines of the power plant system or configuration of the aircraft. As previously noted, when electrical or other types of augmentation systems are utilized, the total equivalent power required by the aircraft for performance comparison is simple to determine. For comparison or evaluation of preliminary designs, the estimated fuel flow requirements could be utilized in the same manner. Although estimated fuel consumption rates might be no more accurate as a basis for comparison than estimated power requirements, they would be as convenient. In addition, the use of fuel flow rates would yield a performance comparison based on total aircraft efficiency.

3. Since range, endurance, payload, and cost of operation of V/STOL vehicles are all directly related to fuel consumption, indications of the performance capabilities for aircraft having various gross weights, power plant systems, or aerodynamic configurations could be obtained from efficiency parameters based on fuel consumption rates. For example, the use of fuel consumption rates to establish a new definition of the equivalent lift-to-drag ratio based on thermal power has been proposed by Stepniewski (Reference 54). Also, Toms (Reference 58)

has suggested the use of dimensionless "numbers", using Q as a variable, to identify flow regimes for V/STOL aircraft having lift or thrust augmentation by power.

SUMMARY OF TECHNIQUES AND METHODS OF PERFORMANCE EVALUATION

After a preliminary analysis of the various techniques and methods discussed above, it was concluded that the use of thermal energy provided a common basis for evaluation and comparison of the overall performance capabilities of V/STOL aircraft. Consideration of the total mechanical energy technique led to the conclusion that this method should be employed to optimize the flight schedules of V/STOL aircraft in cases where the improvement of performance due to the optimization process is expected to be significant. This decision to optimize flight schedules would be based on the performance capability of the aircraft, the cruise altitude and velocity, and the degree of accuracy to which the performance of the aircraft must be determined. Generalized performance techniques appeared to be as applicable to V/STOL aircraft as to fixed-wing aircraft in level flight. The generalized performance method was not utilized in this work, however, since it was of interest to illustrate the actual power and thrust requirements of aircraft as a function of influencing variables. It was further concluded that dimensional analysis should be used to determine new dimensionless parameters which would include fuel flow rate as a new variable.

The aircraft selected for this analysis were typical V/STOL aircraft for which performance data were available. Because of the scarcity of climbing and level-flight performance data for high-speed jet V/STOL aircraft, further analysis of total mechanical energy methods for optimization

of flight profiles was impossible. For this reason, full attention was given to the use of equivalent thermal energy methods of evaluating and comparing the performance of V/STOL aircraft. This work is presented in Phases 1 and 2 of this report. Dimensionless parameters based on fuel flow rate are discussed in Phase 3.

WORK PERFORMED

A literature survey was made to determine the currently used methods of performance evaluation for V/STOL aircraft and to examine the degree of applicability of these methods to other types of vehicles. In addition, visits to aircraft companies and government agencies were made for conferences with personnel involved in V/STOL research, design, and flight testing. Performance data were obtained whenever possible for use in the V/STOL performance and evaluation study, which was performed in the following three phases:

Phase 1. Comparisons were made of a number of V/STOL aircraft of different configurations in level flight. The purpose of these comparisons was to determine and illustrate the relative performance capabilities of each type of aircraft as indicated by parameters currently used for both conventional fixed-wing aircraft and helicopters. New parameters using fuel flow rate as a variable were also studied. Thermal and propulsive efficiencies were examined for the various configurations when sufficient data were available.

Phase 2. A more detailed analysis of the performance capabilities of the following aircraft was performed: a fixed-wing, deflected slipstream-type (OV-10A); a tilt-ducted propeller-type (X-22A); and a conventional helicopter (UH-1B). These aircraft were chosen because of their different configurations and diverse performance capabilities. This phase was devoted to an examination of the effects of gross weight, fuel-to-payload ratio, and altitude on the aerodynamic performance of each type of vehicle as indicated by performance parameters based on fuel flow, payload, and power requirements.

Phase 3. A parametric study was made to investigate the use of nondimensional parameters to predict the aerodynamic forces acting on aircraft deriving high-lift characteristics by power augmentation. Several nondimensional numbers were analyzed, and the validity of their use for particular flight regimes and aircraft was considered. The use of these numbers to predict full-scale performance from model data and for comparison of full-scale aircraft was also included in this phase.

The scope of this investigation and the achievement of its intended objectives were limited to a large extent by several factors. One of these was the difficulty of acquiring a sufficient amount of data for currently operational or planned V/STOL vehicles in the time allotted for this study. In addition to this problem it was found that available data for the hover, transition, and climbing regimes of flight were insufficient to allow a comprehensive overall comparison of V/STOL performance. As a result, it was necessary to confine the study primarily to the level flight condition.

Another difficulty encountered in making performance comparisons of aircraft from published reports was a result of the nonstandard methods of data presentation and notation used by aircraft companies and other agencies. Usually, data for fixed-wing aircraft and helicopters are presented in a standard form; however, the notation used frequently differs. For the newer types of V/STOL aircraft, standards have not been established, and data are generally presented in the most convenient manner for the particular aircraft involved or according to established company procedure. In addition, parameters describing aircraft performance will often contain variables which apply only to a particular vehicle and not to other types of aircraft. As a result, performance comparisons of different types of V/STOL aircraft

were often found to be difficult and time-consuming.

The comparisons of aircraft presented in this report are intended to illustrate only the relative merits and flight regimes of different types of aircraft as indicated by various performance parameters. Since the information used was largely obtained from preliminary performance reports, the data should be considered only as estimated data except in the cases where flight-test results were used (see references). Fuel flow rates, when presented as being 5-percent conservative, were corrected to actual estimated values.

PHASE 1. RELATIVE LEVEL-FLIGHT PERFORMANCE COMPARISONS OF V/STOL AIRCRAFT

A number of aircraft of the STOL or VTOL type are compared in Figures 5 through 21. The objectives of this comparison phase were:

1. To determine the relative range, endurance, and payload capabilities of each type of aircraft for typical power and gross weight conditions.

2. To relate the performance of each aircraft to its thermal, propulsive, and overall aerodynamic efficiencies.

3. To compare the relative performance of these aircraft on the basis of both power required and fuel consumption rates.

Table I lists the aircraft and configurations compared in this phase. The UH-2 and UH-1 compound helicopters were included in this analysis only where sufficient data were available.

COMPARISON OF AIRCRAFT BASED ON FUEL FLOW RATES

Figure 5 shows the rate of fuel consumption Q for several aircraft differing in configuration and gross weight. Curves for the X-22A, XC-142A, and Model 48 Charger are from estimated data. This comparison shows the large variation in fuel consumption rates that exists among the various aircraft and illustrates the rapid rise of fuel consumption for the fixed-wing YHC-1A as airspeed increases. It should be noted that the YHC-1A is powered by a reciprocating engine, whereas all other aircraft in Figure 5 are turbine-powered. It is also significant to note the speed ranges of the various types of aircraft relative to fuel flow required. The limited speed capability of the helicopter is clearly

illustrated by comparing the UH-1B and K-600-3 helicopters with the Charger for approximately the same gross weight. Although these aircraft use approximately the same amount of fuel, the penalty paid for VTOL by the helicopter is a restriction of speed of almost 2.5 times that of the STOL Charger.

It is much more meaningful to examine the fuel flow rate required by each aircraft relative to total aircraft weight. Figure 6 shows the relative fuel flow rates in terms of Q/W, the inverse of specific impulse. Since Q is a function of thermal, mechanical, and aerodynamic efficiency, Q/W may be considered as an inverse measure of overall aircraft efficiency. In fact, an equivalent lift-to-drag ratio based on fuel flow rate may be expressed in terms of specific impulse, W/Q. This can be accomplished by expressing Q in terms of thermal power by the use of fuel heat values.

$$Q \times q \times \frac{3.93}{10^4} = (HP)_T \qquad (5)$$

where $(HP)_T$ is thermal horsepower.

In this report, a fuel heat value of 18,500 q was assumed for the fuel used by turbine or turbojet-powered aircraft, and 19,000 q was assumed for the fuel used by the reciprocating engines of the YAC-1DH Caribou. It follows that

$$(HP)_T = Q \times 18{,}500 \times \frac{3.93}{10^4} = 7.2705\ Q \text{ (for turbine engines)} \qquad (6)$$

and

$$(HP)_T = Q \times 19{,}000 \times \frac{3.93}{10^4} = 7.4670 \text{ (for reciprocating engines)} \qquad (7)$$

Since total lift is equal to weight and drag is equal to thrust for equilibrium level flight, we may express the lift-to-drag ratio of an aircraft as

$$\frac{L}{D} = \frac{W}{T} = \frac{(W \times V)/325.5}{(T \times V)/325.5} = \frac{WV}{325.5\ (THP)} \tag{8}$$

Here, L/D expressed in terms of thrust horsepower (THP) is a relation indicative of aerodynamic efficiency. By replacing THP with thermal power $(HP)_T$, a lift-to-drag ratio is obtained which is indicative of the total efficiency of an aircraft. In terms of W/Q, we have

$$\left(\frac{L}{D}\right)_T = \frac{WV}{325.5\ (HP)_T} = \left(\frac{W}{Q}\right)\left(\frac{V}{325.5 \times 7.27}\right) \cong \frac{WV}{2366\ Q} \tag{9}$$

for turbine-powered aircraft. For reciprocating engines,

$$\left(\frac{L}{D}\right)_T \cong \frac{WV}{2430\ Q} \tag{10}$$

Lines of $(L/D)_T$ values (Equation 9) are shown in Figure 6. This figure shows the fixed-wing YHC-1A as having the lowest relative fuel consumption and highest overall lift-to-drag ratio, as might be expected. It also indicates that although the relative fuel flow rate for helicopters is approximately the same as for the Charger and X-22A with two operating engines, the helicopters operate at lower lift-to-drag ratios; thus, they have lower overall efficiencies. Figure 6 shows the tilt-wing XC-142A to be the most efficient V/STOL aircraft when operating on two engines; it shows the deflected-thrust X-14 (an earlier research vehicle) to be the poorest.

It is also of interest to compare the YHC-1A helicopter

and X-22A with four operating engines. Although both aircraft operate at approximately the same overall efficiency, the relative fuel flow rate of the X-22A is higher, and thus the X-22A is more expensive to operate from a fuel economy standpoint. As a result, the YHC-1A would generally be chosen for low-speed economical operation and the X-22A for missions where higher airspeed is required.

While the above comparison describes the overall efficiency of aircraft, it does not indicate the payload capability of these aircraft. Thus, it is useful to examine the fuel flow rate relative to the payload weight of each vehicle. One may also redefine the lift-to-drag ratio in terms of that portion of lift that is equal to the payload in order to determine what could be called a "payload efficiency" parameter,

$$\left(\frac{L}{D}\right)_u \cong \frac{W_u V}{2366\ Q} \text{ (for turbine engines)} \tag{11}$$

Fuel flow rate relative to payload is shown in Figure 7, along with lines of constant $(L/D)_u$. On the basis of payload capability, aircraft having high rates of fuel consumption, but which are capable of transporting heavy payloads, will compare favorably with those aircraft having lower fuel consumption requirements and payload capability. As illustrated by the UH-1B in Figure 7, a reduction of aircraft weight at the expense of payload will result in a lower value of payload efficiency, since the reduction of fuel required will not be directly proportional to the reduction of payload. This type of comparison is most useful when evaluating aircraft such as heavy-lift helicopters or transport aircraft, providing range and endurance are of secondary importance.

Since the above comparison considers only the ability

of aircraft to carry a load and does not consider the fuel load, the analysis can be still further extended to allow a comparison on the basis of total fuel and payload capability. This type of comparison is necessary when both payload and range requirements are important. Figure 3 illustrates this type of comparison, where

$$\left(\frac{L}{D}\right)_p \cong \frac{(W_u + W_f)V}{2366\ Q} \text{ (for turbine engines)} \tag{12}$$

This type of comparison shows that the XC-142A has the best payload-range trade-off capability (operating on two engines) of those aircraft compared. This indicates that for a given payload, the XC-142A will be capable of flying greater distances than the other aircraft. This comparison is similar to that of Figure 6, except that the empty weight of the aircraft plus the weight of the crew, nonusable fuel, etc., has been eliminated. A comparison of aircraft in this manner allows the weight penalty paid for V/STOL operation to be included in the analysis.

COMPARISCN OF THERMAL AND EQUIVALENT SHAFT HORSEPOWER REQUIREMENTS OF AIRCRAFT

The performance of aircraft is commonly measured relative to the shaft horsepower required in flight. This method of evaluation neglects the thermal efficiency of the aircraft engines and does not permit a performance evaluation with respect to total or overall efficiency. In cases where thermal efficiency of the different aircraft being compared is the same, an evaluation based on shaft horsepower is sufficient to allow a meaningful comparison of the aircraft involved. When thrust engines are used in combination with other power plants driving propellers, fans, or rotors, an equivalent shaft horsepower is frequently defined as

$$SHP_e = SHP + \frac{T \times V}{325.5\ \eta_{pr}} \tag{13}$$

Figure 9 shows the equivalent shaft horsepower required for the aircraft previously compared in Figure 5, as well as that required for two compound helicopters - the UH-1 and UH-2 compounds. An indication of the relationship between the fuel flow rate and the equivalent shaft horsepower required may be obtained by comparing Figure 9 with Figure 5. The plots are similar, but the relative position of the curves has changed in some instances because of differences in thermal efficiencies of the various aircraft. A more significant comparison may be obtained from the relative thermal power curves of Figure 10 and the equivalent horsepower-required curves of Figures 11 and 12.

In general, reducing the number of operating engines results in a decrease of relative thermal power due to increased thermal efficiency. Figure 10 shows an increase in overall efficiency of 35 percent for the X-22A at 160 knots when the aircraft is operating on two engines and four propellers. Figure 11 shows that approximately 40 percent of this increase is due to increased propulsive efficiency.

Comparisons may be made at given velocities to determine the difference in aircraft efficiency as indicated by relative shaft horsepower and thermal power. On the basis of relative shaft horsepower, for example, the XC-142A at 200 knots is 41 percent more efficient when operating on two engines at 37,907 pounds than the Model 48 Charger at the same velocity and weighing 8350 pounds. On the basis of thermal power, however, the XC-142A is shown to be 45 percent more efficient than the Charger due to the slightly better thermal efficiency of the XC-142A. Similarly, the UH-1B, K-600-3, and YHC-1A helicopters appear to be equally

efficient at approximately 80 knots airspeed when evaluated with respect to relative shaft horsepower. When compared using relative thermal power, the UH-1B is 32 and 10 percent more efficient than the K-600-3 and YHC-1A, respectively, for the aircraft weights shown in Figures 10 and 11.

In comparison to V/STOL aircraft, the fixed-wing YAC-1DH Caribou has a minimum relative shaft horsepower rating of approximately 0.026 and a relative thermal power rating of about 0.110. This is 40 percent better than the XC-142A operating on two engines when relative shaft horsepower is used as a basis for comparison, and 44 percent better when relative thermal power is used for comparison. It should be noted, however, that the XC-142A becomes as efficient as the Caribou at 160 knots airspeed, and it is actually a more efficient aircraft above this speed.

The relative shaft horsepower curves for the UH-1 and UH-2 compounds are shown in Figure 12. These curves indicate a speed advantage of approximately 30 knots for the advanced UH-1 compound with wings when compared at equal power-to-weight values over the speed range of these aircraft. Comparison at a given airspeed shows the UH-1 compound to be about 35 percent more efficient than the UH-2 compound without wings.

LIFT-TO-DRAG RATIOS BASED ON THERMAL AND EQUIVALENT SHAFT HORSEPOWER

Values of constant lift-to-drag ratio based on equivalent shaft horsepower are plotted on Figures 11 and 12. Equivalent lift-to-drag ratio is defined as

$$\left(\frac{L}{D}\right)_e = \frac{\eta_{pr} WV}{325.5 \text{ THP}} = \frac{WV}{325.5 \text{ SHP}} \tag{14}$$

For aircraft such as compound helicopters which use auxiliary jets for increased thrust,

$$\left(\frac{L}{D}\right)_e = \frac{WV}{325.5\ SHP_e} \tag{15}$$

Since the expression for equivalent lift-to-drag ratio includes the propulsive efficiency, $(L/D)_e$ is a measure of the efficiency of an aircraft which excludes the thermal efficiency of the power plants. From Figures 11 and 12, one may observe the relationship between relative power and equivalent lift-to-drag ratio. A comparison of the YAC-1DH Caribou and XC-142A with two operating engines shows that these two aircraft operate within the same range of $(L/D)_e$ values although the relative power required for the XC-142A is larger than that of the Caribou. This occurs since the higher power requirements of the XC-142A are realized as increased velocity capability.

Maximum values of lift-to-drag ratio may be determined from Figures 13 and 14. From these figures, a comparison may be obtained of lift-to-drag ratios based on both equivalent shaft horsepower and thermal heat. Included for comparison purposes are two additional aircraft, the STOL 941 and the CL-84A (Figure 14).

A comparison of lift-to-drag ratios for the X-22A illustrates the improved efficiency of this aircraft when operating on two engines with four propellers as compared with four-engine operation. Overall efficiency is increased by 33 percent, as indicated in Figure 13 by maximum values of $(L/D)_T$. Figure 14 shows that a small percentage of the increased overall efficiency is due to increased propulsive efficiency. Since in both cases the aircraft is using four propellers, this small change in propulsive efficiency may

be attributed to the increased mechanical efficiency of the aircraft.

Figure 13 also shows that the Model 48 Charger has a higher thermal lift-to-drag ratio when operating at its highest weight condition. Figure 14 shows the opposite result when only propulsive efficiency is considered. This situation occurs because of the inverse relationships of propulsive and thermal efficiencies with weight for this aircraft. This shows that when aircraft are evaluated using equivalent lift-to-drag ratio, the aircraft appearing to be most efficient may not be the most economical to operate or may not have the best range characteristics because of low thermal efficiency and a correspondingly higher rate of fuel consumption.

A comparison of maximum equivalent lift-to-drag values from Figure 14 also shows that the fixed-wing YAC-1DH is 11 percent more efficient than the XC-142A with two operating engines. However, the high thermal efficiency of the YAC-1DH at lower airspeeds results in a difference of 58 percent when compared with the XC-142A in Figure 13. The relatively high equivalent lift-to-drag ratio of the XC-142A indicates good propulsive efficiency for this tilt-wing aircraft, $(L/D)_e$ values being slightly greater than the deflected-slipstream 941 and about 15 to 20 percent larger than that of the CL-84A at cruise velocities. The curves for the UH-1 and UH-2 compound helicopters show that the price paid for increased airspeed is a rapid decrease in equivalent lift-to-drag ratio. However, it is interesting to note the improvement of the lower-drag advanced UH-1 compound with wings over the UH-2 compound. Nevertheless, at 200 knots airspeed, the lift-to-drag ratio of the UH-1 compound is still only 71 percent of that for the tilt-duct X-22A with four operating engines.

THERMAL, PROPULSIVE, AND TOTAL EFFICIENCIES OF V/STOL AIRCRAFT

The thermal efficiency of an aircraft power plant is defined as the ratio of output shaft horsepower to thermal power.

$$\eta_T = \frac{SHP}{(HP)_T} = \frac{THP}{\eta_{pr} \times (HP)_T} \qquad (16)$$

When aircraft have thrust engines or other auxiliary sources of power, η_T may be defined in terms of equivalent shaft horsepower and thermal power calculated from the total rate of fuel consumption of the aircraft as

$$\eta_T = \frac{SHP_e}{[(HP)_T]_{total}} \qquad (17)$$

Figure 15 shows the increasing thermal efficiency of turbojet and turboprop aircraft at high forward speeds. The YAC-1DH, powered by reciprocating engines, also exhibits increasing thermal efficiency up to approximately 140 knots, after which the thermal efficiency decreases rapidly. Even so, the thermal efficiency of the reciprocating engines of this aircraft is relatively good at airspeeds up to approximately 165 knots.

A comparison of thermal efficiencies of V/STOL aircraft is important in evaluating the aircraft for range, endurance, and payload capability, since higher thermal efficiency means greater fuel economy, longer range, and increased endurance. Figure 15 illustrates the increased thermal efficiency obtained with two-engine operation of the X-22A and the generally lower thermal efficiencies of the helicopters and pure jet X-14 aircraft.

While the thermal efficiencies of the Model 48 Charger,

X-22A, and XC-142A are quite close for the two-engine operation, as indicated in Figure 15, the X-22A suffers a loss of propulsive efficiency with increasing airspeed while the propulsive efficiencies of the other aircraft remain high. The propulsive efficiencies of these aircraft are shown in Figure 16, where

$$\eta_{pr} = \frac{THP}{SHP} = \frac{D \times V}{325.5 \ SHP} \tag{18}$$

Figure 16 shows the effect of the increasing drag of the ducted-propeller system of the X-22A with increasing airspeed. Propulsive efficiency of the X-22A is comparable to that of the Charger and XC-142A at airspeeds near 120 knots, but it steadily decreases with increasing velocity. This effect, coupled with increasing thermal efficiency, gives this aircraft a rather flat overall efficiency curve over its speed range, as illustrated in Figure 17, where

$$\eta_o = \eta_{pr} \times \eta_T \tag{19}$$

The higher overall efficiency of the XC-142A and Model 48 Charger is indicative of the low relative fuel and shaft horsepower requirements and payload capabilities previously discussed. The very low efficiency of the pure jet X-14 illustrates the superiority of turboprop propulsion for the airspeeds considered.

RANGE AND ENDURANCE CHARACTERISTICS

It is important to examine the ratio of velocity-to-fuel rate Q, or specific range, to determine range capability. Figure 18 shows the specific range of various aircraft at given aircraft weights. From this plot, the velocity for maximum range may be determined at maximum values of specific

range. Since specific range is a function of fuel flow rate and, therefore, also a function of aircraft weight, Figure 18 shows only the relative ratings of specific range for the various types of aircraft as determined from the aircraft weights shown.

Figure 19 shows the range of these aircraft with usable fuel loads as given in Table I. In this plot, the general range characteristics of the various configurations are illustrated - the low-speed superiority of the piston-engine YAC-1DH, the long-range capability of the tilt-wing XC-142A and deflected-slipstream Charger, and the relatively short-range characteristics of the helicopters and turbojet-powered X-14.

Endurance characteristics of these aircraft are plotted in Figure 20. It should be emphasized that fuel consumed for takeoff and climb was not considered in this level-flight comparison.

SUMMARY OF PHASE 1 EVALUATION

Table II shows the important efficiency and performance characteristics for some of the aircraft configurations used in the preceding comparison and evaluation phase. The values shown in Table II are for the aircraft weights and fuel loads which were considered in the preceding analysis and which may be found in Table I. The loading conditions chosen were typical of mission loading conditions or were those of the aircraft in a flight-test configuration. Although values in Table II are not to be considered optimum ones, a general picture of the relative capabilities of the V/STOL types considered may be obtained.

Comparisons of lift-to-drag ratio based on thermal and shaft horsepower in Table II indicate that these values are generally consistent with range and endurance capability. The relationship between thermal lift-to-drag ratio and

TABLE 1

AIRCRAFT CONFIGURATIONS AND DATA

Phase 1 Comparisons

Aircraft Designation	Type	Installed Power	Mission or Configuration	Gross Weight (lb)	Payload (lb)	Fuel load (lb)	No. Engines and Props
Model 48 "Charger" (Refs. 11, 12, 14)	STOL Deflected Slipstream	Two T-74 CP-8 Turboprops 565 SHP Each at 2200 rpm (NRP)	A. Visual Reconnaissance	6,262	0	1,292	2 Engines 2 Props
			B. Close Air Support	8,350	2,220	870	2 Engines 2 Props
XC-142A (Ref. 15)	V/STOL Tilt Wing	Four T-64 GE-1 Turboprops 2700 SHP Each (MRP)	A. 200-N.-Mile Radius	37,907	8,000	5,648	2 Engines 4 Props
			B. 100-N.-Mile Radius	35,668	8,000	3,409	2 Engines 4 Props
X-22A (Ref. 52)	V/STOL Tilt Ducted Prop	Four YT58-GE-80 Engines - 1050 SHP Each (NRP)	A. Endurance	15,930	1,200	3,162	4 Engines 4 Props
			B. Range	14,833	1,200	2,065	4 Engines 4 Props
			C. Endurance	15,930	1,200	3,162	2 Engines 4 Props
			D. Range	14,833	1,200	2,065	2 Engines 4 Props
UH-1B (Ref. 27)	Pure Helicopter	One T-53-L-11 Engine - 900 SHP (NRP)	A. Basic Mission	7,085	750	1,573	1 Engine Rotor
			B. Internal Cargo	9,500	2,763	1,573	1 Engine Rotor
YAC-1DH "Caribou" (Ref. 16)	Fixed Wing	Two R-2000-7M2 Reciprocating Engines 1200 BHP Each (NRP)	Flight Test	26,000	2,988	4,200	2 Engines 2 Props
K-600-3 (Ref. 47)	Dual Rotor Helicopter	One YT53-L-1 Turbine-600 SHP (NRP)	Flight Test	6,630	1,100	970	1 Engine 2 Rotors
YHC-1A (Ref. 19)	Tandem Helicopter	Two T-58-GE-6 Turbines - 1050 SHP Each (NRP)	Flight Test	14,000	2,685	2,240	2 Engines 2 Rotors
X-14 (Ref. 5)	Vectored Thrust V/STOL	Two Viper Turbojets - 1750 lb Thrust Each	A. Flight Test	2,960	0	600	2 Engines
			B. Flight Test	2,960	0	600	1 Engine
UH-2 (Modified) (Ref. 62)	Compound Helicopter	One Primary T58-8B Engine, One Auxiliary YJ-85-5	Flight Test	8,900	0	-	1 Primary Engine, Rotor 1 Auxiliary Jet
UH-1 (Modified) (Ref. 59)	Compound Helicopter	One Primary T53-L-11 - 900 SHP (NRP) Two J69-T29 Turbojet Aux. Engines - 1700 Lb. Thrust (MRP)	Flight Test	9,800	0	-	1 Primary Engine, Rotor 2 Auxiliary Jets
941 (Ref. 56)	STOL Transport		Flight Test	37,800	-	-	4 Engines 4 Props
CL-84A (Ref. 56)	V/STOL Tilt Wing		-	10,500	-	-	2 Engines 2 Props

TABLE II

COMPARISON OF LEVEL-FLIGHT PARAMETERS

Phase 1 Comparisons

Aircraft	$(L/D)_T$ (Max)	(L/D) (Max)	(η_o At	η_{pr} V_{max}	η_T) range	(V/Q) (Max)	V_{max} range (Kn)	V_{max} (NRP) (Kn)	R_{max} (N Mi)	E_{max} (Hr)
YAC-1DH Caribou W_g = 26,000 Lb W_u = 2,988 Lb	2.980	11.17	-	-	0.284	0.2808	118	185	1185	11.75
XC-142A (2 engines) W_g = 37,907 Lb W_u = 8,000 Lb	1.890	10.04	0.164	0.816	0.201	0.1180	188	260	668	4.40
Model 48 Charger W_g = 8,350 Lb W_u = 2,220 Lb	1.298	6.70	0.156	0.788	0.198	0.3680	190	255	320	1.84
Model 48 Charger W_g = 6,262 Lb W_u = 0	1.147	7.80	0.130	0.813	0.160	0.4334	172	260	560	4.08
X-22A (2 engines) W_g = 15,930 Lb W_u = 1,200 Lb	0.956	4.95	0.113	0.565	0.200	0.1424	164	188	448	3.30
X-22A (4 engines) W_g = 15,930 Lb W_u = 1,200 Lb	0.720	4.78	0.084	0.506	0.166	0.1067	180	259	337	2.40
YHC-1A W_g = 14,000 Lb W_u = 2,685 Lb	0.720	4.10	-	-	0.185	0.1221	118	145	273	2.88
UH-1B W_g = 9,500 Lb W_u = 2,763 Lb	0.708	5.14	-	-	0.189	0.1760	102	111	277	3.28
K-600-3 W_g = 6,630 Lb W_u = 1,100 Lb	0.510	3.83	-	-	0.154	0.1850	94	94	180	2.55
X-14 (1 engine) W_g = 2,960 Lb W_u = 0	0.172	1.36	0.029	0.204	0.142	0.1367	115	-	82	0.90

maximum range for typical fuel loads is further illustrated in Figure 21.

In general, the parameters of Table II show that the tilt-wing XC-142A with two operating engines is superior to the other V/STOL aircraft in range and endurance capability and exhibits the highest lift-to-drag ratio based on both thermal and shaft horsepower. In addition, Table II indicates that the X-22A, for both the two- and four-engine operation, is a faster and longer range vehicle than the helicopters that were considered; however, the X-22A is payload deficient by virtue of its lower payload-to-gross-weight ratio. The Model 48 Charger exhibits high specific range characteristics which give this aircraft good range efficiency at relatively high cruise velocities.

Such comparisons of range, velocity, endurance, and payload capability as those above are sufficient only to establish, in a broad sense, the performance characteristics of the various types of V/STOL aircraft. A more detailed analysis in which performance parameters are optimized by trade-offs between fuel and payload and by gross weight adjustments must be performed to allow selection of a particular aircraft for a specific mission.

Further examination of the efficiency parameters and lift-to-drag ratios of Table II reveals that differences of propulsive efficiency, η_{pr}, were considerably larger than those of thermal efficiency, η_T, among the various aircraft examined. Propulsive efficiencies at velocity for maximum range varied from 0.816 to 0.204 as compared to a corresponding variation of thermal efficiency of 0.201 and 0.142. Thus, performance capability will generally be more strongly related to propulsive efficiency, especially when the power plant systems of the aircraft being compared are similar.

The preceding comparison also illustrates the differences

in relative efficiency as indicated by comparisons based on fuel flow rate as opposed to those based on power required. Thermal lift-to-drag ratios and overall efficiency values may in some cases show one configuration to be more efficient than another; whereas an opposite evaluation may be obtained when only equivalent lift-to-drag ratios and propulsive efficiencies are considered. This condition is illustrated in Table II by the values of lift-to-drag ratio and efficiency for the Model 48 Charger. For this reason, it would appear desirable to compare aircraft by use of thermal lift-to-drag ratios and overall efficiency parameters whenever rates of fuel consumption or thermal efficiencies are significantly different. This will also provide a convenient means of accounting for the fuel used by the main power plant system or any auxiliary system to augment performance in any manner.

PHASE 2. INVESTIGATION OF THE PERFORMANCE CAPABILITIES OF THREE V/STOL AIRCRAFT OF DIFFERENT GROSS WEIGHTS AND CONFIGURATIONS AND METHODS OF PERFORMANCE COMPARISON

The preceding comparisons of V/STOL aircraft performance and efficiency did not consider such factors as the change in performance with altitude or the relationship of payload, fuel load, and total weight to the optimization of performance parameters. Also, only level-flight performance was discussed.

The purpose of Phase 2 was to examine the effect of altitude and loading conditions on aircraft performance and to devise parameters which could be used for a comprehensive comparison of V/STOL performance. Efficiency parameters were examined with regard to mission requirements, and a comparison of landing, takeoff, and climb characteristics was made to the extent that available data permitted.

The three aircraft analyzed in this phase were the UH-1B, the OV-10A, and the X-22A. The UH-1B was selected since it is representative of a typical helicopter of current design and because of its true rotary-wing configuration. The OV-10A was chosen as a typical STOL aircraft, and the X-22A was selected because of its larger size and more complex design. Since the analysis was intended to examine methods of performance comparison which are independent of aircraft configuration, the three aircraft were selected only for illustration of the relative performance capabilities as indicated by the parameters investigated.

Table III lists 9 loading conditions for the aircraft used in this analysis. All aircraft weights considered lie within the range between the heaviest and lightest mission gross weights specified by the manufacturer. Only the clean configurations without external cargo, pylons, or racks were considered. Also, only internal usable fuel was considered.

TABLE III

AIRCRAFT LOADING CONDITIONS (OV-10A, X-22A, and UH-1B)

Phase 2 Comparisons
(Internal Fuel Only)

Condition	OV-10A*	X-22A#	UH-1B+
Sea Level Maximum Mission Weight (Lb)	Max. Close Air Support 10,742	Max. Overload 17,706	Cargo 9,500
Sea Level Minimum Mission Weight (Lb)	Visual Recon. 7,761	VTOL Missions 14,833	Trainer 6,869

	OV-10A	X-22A	UH-1B
1. Max. Gross Wt., Max. Fuel and Payload, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,456	3,128	1,573
Payload (Lb)	2,981	2,976	2,583
Gross Weight (Lb)	10,742	17,706	9,500
2. 90% Max. Gross Wt., Max. Payload, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	382	1,357	623
Payload (Lb)	2,981	2,976	2,583
Gross Weight (Lb)	9,668	15,935	8,550
3. 80% Max. Gross Wt., Equal Payloads, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,456	1,730	1,423
Payload (Lb)	833	833	833
Gross Weight (Lb)	8,594	14,165	7,600
4. Maximum Fuel, Zero Payload, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,456	3,128	1,573
Payload (Lb)	0	0	0
Gross Weight (Lb)	7,761	14,730	6,917
5. W_f/W = 0.15, W_u/W = 0.19, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,433	2,637	1,215
Payload (Lb)	1,815	3,341	1,538
Gross Weight (Lb)	9,553	17,580	8,097
6. W_f/W = 0.10, W_u/W = 0.12, S.L.			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	808	1,487	685
Payload (Lb)	970	1,785	822
Gross Weight (Lb)	8,083	14,874	6,851
7. Maximum Fuel, Zero Payload, 10,000 Ft			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,456	3,128	1,573
Payload (Lb)	0	0	0
Gross Weight (Lb)	7,761	14,730	6,917
8. W_f/W = 0.18, W_u/W = 0.04, 10,000 Ft			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	1,455	2,678	1,233
Payload (Lb)	323	596	274
Gross Weight (Lb)	8,083	14,874	6,851
9. W_f/W = 0.10, W_u/W = 0.12, 10,000 Ft			
Basic Weight (Lb)	6,305	11,602	5,344
Usable Fuel (Lb)	808	1,487	685
Payload (Lb)	970	1,785	822
Gross Weight (Lb)	8,083	14,874	6,851

*Ref. 8
#Ref. 52
+Ref. 27

NOTE:
Basic Weight = empty weight plus crew and equipment, trapped fuel and oil, armor, sponsons, pylons, racks, and other noncargo-type equipment
Payload = cargo, ammunition, guns, bombs, rockets, litters, or any disposable-type cargo

The definitions of basic weight and payload as used in this analysis are shown in Table III.

EFFECT OF TOTAL AIRCRAFT WEIGHT AND ALTITUDE ON RELATIVE FUEL FLOW AND POWER REQUIREMENTS

The rate of fuel consumption of the X-22A tilt-duct aircraft is approximately three times that of the UH-1B and OV-10A aircraft (Figures 22 and 23). This is an indication of the rather low propulsive efficiency of the X-22A, for even on a basis of total weight (Q/W), the relative fuel flow rate for this aircraft is over twice that of the OV-10A at sea level (Figure 24). An examination of thermal lift-to-drag ratios shows that the overall efficiency of the X-22A is slightly better than that of the helicopter but only half that of the OV-10A. Since there are only small differences in payload of the three aircraft for the maximum gross weight condition (Condition 1), the above observations may be interpreted to mean that the X-22A achieves a slightly higher thermal lift-to-drag ratio than the helicopter at the expense of a poorer payload-to-total weight ratio. Also, one might consider the differences in relative fuel consumption and thermal L/D ratio between the X-22A and OV-10A as the price paid for VTOL while maintaining the same speed range and payload capability.

Figure 24 reveals the change in fuel requirements with aircraft weight. For each aircraft, the fuel flow rate per pound of aircraft increases as aircraft weight is reduced. A 20-percent reduction in weight of each aircraft resulted in an increase of relative fuel flow rate for the OV-10A, UH-1B, and X-22A of approximately 16, 13, and 18 percent, respectively, at sea level cruise velocities. Thus, operation of any of the three types of aircraft at less than maximum gross weight results in a reduction of fuel economy which is quite significant.

The thermal efficiency of turbine engines increases with altitude. Figure 25 illustrates the large reduction of relative fuel flow rate with a change of 10,000 feet of altitude for the maximum-fuel, zero-payload condition. For instance, at 190 knots, the relative fuel consumption of the X-22A is 24 percent less at 10,000 feet than at sea level for a total weight of 14,730 pounds. The corresponding reductions of Q/W for the OV-10A and UH-1B at 90 knots are 19 and 21 percent, respectively. Therefore, any comparison of the fuel economy of various aircraft must include a thorough analysis of altitude fuel requirements.

High relative fuel flow rate is a severe penalty to the performance of any aircraft, especially a VTOL type because of the large fuel requirements in the vertical, transitional, and hover regimes of flight. The high fuel requirement of the tilt-duct X-22A is further confirmed from a comparison of hover characteristics of the X-22A and XC-142A, both using four engines and propellers (Figure 26). Although the XC-142A is a heavier aircraft, its relative hover fuel rate requirements are indicated as being less than half those of the X-22A. This is surprising when one considers the high static thrust characteristics of ducted propeller designs. It appears that the relative weight penalty of the tilt-duct design is considerably larger than that of the conventional tilt wing.

It is important to note that Figures 27 and 28, which are plots of the relative shaft horsepower required for the same aircraft and conditions illustrated in Figures 24 and 25, do not indicate the same relative changes between aircraft that were shown from the comparisons of relative fuel flow rates. The curves of Figure 27 indicate equal relative power requirements for the X-22A at 180 knots, and Figure 24 shows relative fuel flow rate to be 15 percent less at maximum gross weight at that airspeed. This illustrates the

importance of comparing aircraft on the basis of relative fuel flow rate requirements, since relative power comparisons do not provide a true picture of aircraft performance due to the neglect of thermal efficiency considerations. The use of fuel flow rate is also required to indicate the differences of fuel consumption with changes in altitude due to changes of thermal efficiency.

VARIATION OF THERMAL AND EQUIVALENT LIFT-TO-DRAG RATIO WITH ALTITUDE AND GROSS WEIGHT

It has been shown that the thermal lift-to-drag ratio may be expressed in terms of specific range as

$$\left(\frac{L}{D}\right)_T = \frac{W}{q}\left(\frac{V}{Q}\right) \tag{20}$$

where q is the fuel heat value of the particular fuel used by an aircraft. Figure 29 shows that specific range, V/Q, is reduced by increasing aircraft weight. Figure 30, which is a plot of maximum specific range values as a function of total weight, shows the downward trend of V/Q as total aircraft weight is increased. It should be noted that the product of total weight and specific range is largest at the highest weight condition for each aircraft. As a result, the thermal lift-to-drag ratio as defined above will be maximum for the maximum gross weight condition of each aircraft.

Figure 31 shows typical variations of $(L/D)_T$ with aircraft weight at sea level. For all three aircraft, a 20-percent decrease in total weight is equivalent to a corresponding decrease of $(L/D)_T$ of about 15 percent. Similar correlations of weight to thermal lift-to-drag ratio should not be expected for all aircraft, however, since $(L/D)_T$ is strongly dependent upon the thermal efficiency of the type

of power plant being considered. The large differences between the thermal lift-to-drag ratios of the STOL OV-10A and the VTOL aircraft as shown in Figure 31 again illustrate the reduced efficiency of the relatively heavier and more mechanically complex VTOL machines.

A comparison of equivalent L/D ratios of Figure 32 shows closer agreement between the OV-10A and X-22A, which is an indication that thermal efficiency of the STOL aircraft is higher than that of the X-22A. The thermal efficiencies of Figure 33 show this to be true. The equivalent lift- to-drag ratio for the OV-10A shows that propulsive efficiency of this aircraft is independent of aircraft weight at a velocity of 158 knots, near maximum $(L/D)_e$. However, Figure 31 shows that a mismatch of maximum propulsive and thermal efficiencies results in maximum $(L/D)_T$ ratios occurring at slightly higher velocities - 170 to 185 knots for the weights considered. The X-22A shows better matching of thermal and propulsive efficiencies at sea level.

Figures 34 and 35 illustrate the large improvements of thermal efficiency and lift-to-drag ratio that occur for all three aircraft for an increase in altitude of 10,000 feet. From Figure 35, it can be seen that the thermal efficiency of the X-22A at 10,000 feet is approaching that of the OV-10A, which is considerably more efficient at sea level.

The above comparisons are intended to show some of the effects of total aircraft weight and altitude on lift-to-drag ratios of the three different types of aircraft used for illustration. The changes of efficiency of the three aircraft with altitude and weight as indicated by the above comparisons will be reflected as changes in range, endurance, and payload capabilities.

SELECTION OF LOADING CONDITIONS AND EFFECT OF ALTITUDE ON RANGE AND ENDURANCE CAPABILITY

The range characteristics of the UH-1B, X-22A, and OV-10A are shown in Figures 36 and 37. Since range is a function of velocity, total weight, fuel load, payload, and altitude, several loading conditions were analyzed at sea level and at 10,000 feet to determine ways in which aircraft of different gross weight and fuel capacity could best be compared on a fair and meaningful basis. The range in nautical miles as shown is optimistic, since the curves represent a hypothetical range for each aircraft that would occur if all internal fuel were expended in level flight.

Range in nautical miles may be expressed as

$$R = W_f \left(\frac{V}{Q}\right), \text{ where V is airspeed in knots} \tag{21}$$

Thus, range is expressed as a function of the fuel load in pounds and the specific range V/Q in nautical miles per pound of fuel. Inspection of Figure 29 shows that V/Q is maximum for minimum overall weight and increases in magnitude with altitude. Since range is also a function of the fuel on board, it can be shown that maximum range will occur for the smallest total weight condition of an aircraft carrying a maximum fuel load.

Figure 36 shows comparisons of range at maximum and minimum aircraft weight conditions with the aircraft carrying a full internal load of fuel (Conditions 1 and 4). Since the minimum weight condition of each aircraft is for zero payload, these curves represent the trade-off of payload for range. For instance, the maximum range of the OV-10A can be increased by 66 miles if the payload of 2981 pounds is sacrificed - an increase in range of approximately 13 percent. For the X-22A and UH-1B, the payload is equivalent to even

smaller percentages of maximum range, since the payload and fuel load-to-total weight ratios for the VTOL vehicles are smaller than for the STOL aircraft.

Since the normal operating weights of the UH-1B, X-22A, and OV-10A are quite different, consideration must be given to a method which will allow a fair comparison of the range capability of each aircraft on a basis of the fuel and payload that each aircraft can carry. This may be accomplished by a comparison of the aircraft having identical fuel and payload-to-total weight ratios. This comparison is illustrated in Figure 36 for a fuel load-to-total weight ratio of 0.15 and a payload-to-total weight ratio of 0.19. Other such ratios may be used as long as the weight of each aircraft remains within the range of normal operation.

A comparison on this basis (Condition 5, Figure 36) shows that the range capability of the X-22A and UH-1B is only 52 and 43 percent of the OV-10A, respectively. At a 10,000-foot altitude. the range capability of the X-22A and UH-1B relative to the OV-10A remains virtually unchanged, as shown by Condition 8 in Figure 37.

The percentage increase of maximum range for an altitude change of 10,000 feet for the maximum-fuel, zero-payload condition was approximately 23, 22, and 31 percent for the UH-1B, OV-10A, and X-22A, respectively. This again shows the more rapid increase of range efficiency of the tilt-duct X-22A with increasing altitude.

Figure 38 shows the endurance characteristics of each aircraft for the maximum-fuel, zero-payload condition and for equal fuel and payload-to-total weight ratios. Endurance, which is equal to W_f/Q, is shown to be maximum for the maximum-fuel, zero-payload condition, as indicated by Conditions 4 and 7. Comparison by equal fuel and payload-to-total weight ratios (Condition 5) shows the relative differences of endurance capability of the three aircraft, the

X-22A being poorer than the other two aircraft. The endurance capability of the UH-1B at 10,000 feet is approximately the same as that of the OV-10A, although velocity for maximum endurance is quite different. Thus, for observation or surveillance missions, altitude and velocity requirements in addition to VTOL capability would determine to a large extent whether the helicopter or STOL OV-10A would be selected for the particular mission being considered.

The plots of maximum range and endurance with variation of payload, shown in Figures 39 and 40, form envelopes from which the relative performance of the aircraft may be easily visualized.

EXAMINATION OF PRODUCTIVITY PARAMETERS FOR V/STOL AIRCRAFT

The selection of the best vehicle for a given mission on the basis of performance is generally achieved by evaluation of certain parameters which express the capability of each aircraft in terms of range, endurance, payload, fuel economy, and velocity, or in terms of combinations of these capabilities. The parameters used for evaluation depend entirely upon the mission requirements. For example, the most important parameters used to evaluate a long-range transport-type aircraft would be those which express the trade-off between range and payload. For a short-range, heavy-lift aircraft such as a crane-type helicopter, payload and endurance would be the most important.

Range and endurance capabilities of certain V/STOL aircraft have previously been illustrated and discussed. The following analysis was made considering those parameters which permit an evaluation and performance comparison of V/STOL aircraft with regard to particular mission requirements. The parameters discussed are all combinations of the following variables: fuel consumption rate (Q), fuel load (W_f), payload (W_u), and airspeed (V). When these variables are known

as a function of aircraft weight (W) and altitude, the productivity parameters can be found. Any aircraft can then be evaluated for its ability to meet mission requirements.

PAYLOAD CAPABILITY

In this report, payload is defined as any type of disposable cargo or armament transported by an aircraft that is not required for flight. Thus, the term is applicable to any aircraft that is used for transporting or carrying guns, ammunition, bombs, litters, troops, or any other kind of disposable load. If the term payload capability is defined by W_uQ/W_f, one can measure the ability of an aircraft to carry a given load per unit of time, relative to some other parameter such as range, endurance, or airspeed.

Figure 41 shows comparisons of the UH-1B, X-22A, and OV-10A for a maximum and a 90-percent maximum gross weight condition as functions of velocity. The parameter W_uQ/W_f is not maximized since this would occur for a minimum fuel load condition. Instead, the payload capability must be measured against range or endurance in addition to velocity, since it is inversely proportional to these quantities.

Figure 41 shows slightly higher values of W_uQ/W_f for the X-22A than for the UH-1B and OV-10A at maximum gross weight with full internal fuel load (Condition 1). However, Figure 42, which gives payload capability versus range, shows the sacrifice of range by the X-22A for this condition. Maximum range of the X-22A is only 63 percent of that of the OV-10A, but it is about 9 percent better than that of the UH-1B. When the gross weights of these aircraft are reduced by 10 percent at the expense of fuel load, payload capability goes up as indicated by Condition 2, but it is accompanied by a corresponding decrease in range. Condition 2, at equal velocity, shows the OV-10A to have a slightly higher payload capability than the X-22A, but slightly less range when

carrying the same payload per hour. The range increase and payload capability decrease with change in altitude are also illustrated by Conditions 6 and 9 in Figure 42.

The trade-off of payload capability for range as shown in Figure 42 allows a comparison of any type of aircraft. For missions where range and payload capability are both of importance, it is convenient to form the parameter W_uR to evaluate the ability of an aircraft to carry a load a given distance. In terms of velocity, the parameter W_uV may be used to express the rate of payload-range capability as in Figure 43. For near-equal payloads, as shown, the X-22A is superior to the OV-10A by virtue of its higher speed at normal rated power. Because of its lower speed capability, the UH-1B helicopter falls far below the other two aircraft. It should be remembered, however, that the expense of operating the much heavier X-22A is paid in terms of high fuel consumption and reduced range.

PAYLOAD-RANGE EFFICIENCY

The payload-range productivity of aircraft may also be evaluated in terms of the fuel used by examining the parameter W_uV/Q. This parameter yields the product of payload and range per pound of fuel consumed by the aircraft. Because of its relation to fuel economy, this parameter is, in effect, an expression of payload and range efficiency.

The parameter W_uV/Q may be maximized by reducing aircraft weight at the expense of fuel. Again, a compromise between payload and range must be made, since a high value of payload-range efficiency for reasonably large payloads is equivalent to reduced range. This is illustrated in Figure 44 by Conditions 1 and 2, which show the increased efficiency of the aircraft with a 10-percent overall weight reduction while keeping the payload constant. By virtue of its higher payload-to-total weight ratio, the OV-10A is

superior when rated on this basis. Even when payload-to-total weight ratio is constant for all three aircraft, as in Condition 6, the OV-10A is still the most capable aircraft. When operating at 10,000 feet, as shown by Condition 8, the UH-1B becomes more payload-range efficient than the X-22A because of its higher rate of increasing thermal efficiency with increasing altitude.

It is not generally profitable to operate an aircraft at reduced weight at the expense of the fuel load. A comparison of the OV-10A, UH-1B, and X-22A for Conditions 1 and 2 in Figure 45 shows that a 10-percent reduction of weight by means of reducing the fuel load resulted in large reductions of endurance but only small increases of payload-range efficiency. For example, an endurance and range reduction of 54 percent occurred for the X-22A for only a 3 percent increase in the payload-range parameter.

The relatively poor payload-range efficiency of the X-22A is illustrated in Figure 46. As a VTOL-type aircraft, a payload-range efficiency equal to that of a STOL aircraft, such as the OV-10A, should not be expected: however, the X-22A is still slightly payload-range deficient when compared to the UH-1B helicopter. Since the maximum range of the X-22A is only about 15 percent greater than that of the UH-1B, selection of the X-22A for missions requiring VTOL capability would depend heavily on velocity requirements.

COMPARISON OF PAYLOAD CAPABILITY RELATIVE TO FUEL FLOW REQUIREMENTS

When aircraft such as cargo or reconnaissance types are designed to carry large payloads or to have high endurance characteristics, the payload-to-fuel flow rate ratio (W_u/Q) is convenient for evaluation and comparison purposes. Since large values of this parameter are indicative of either high payload or low fuel flow, W_u/Q is essentially a payload-

endurance efficiency factor. For the fully loaded, maximum gross weight condition with maximum internal fuel, the OV-10A shows the largest values of this parameter (Condition 1, Figure 47) because of its smaller rate of fuel consumption over the lower half of its speed range. The high rate of fuel consumption and the low payload-to-total weight ratio cause the payload-endurance capability of the X-22A to be small. Figure 48 shows the correspondingly superior range characteristics of the OV-10A as would be expected, and the relatively poor range and payload-endurance rating of the X-22A. Again, one may increase payload-endurance productivity by reductions of gross weight at the expense of fuel, but the improved fuel economy at reduced gross weight does not nearly compensate for loss in fuel in terms of range and endurance reductions, as illustrated by Figure 48.

An increase of both the payload-endurance capability and range of the three aircraft at a 10,000-foot altitude is illustrated in Figure 48 by an evaluation using equal fuel and payload-to-total weight ratios (Conditions 6 and 9).

RANGE-ENDURANCE PRODUCTIVITY

$W_f V/Q^2$, which is formed from the product of endurance (W_f/Q) and specific range (V/Q), provides a measure of the range-endurance capability of an aircraft with respect to the number of pounds of fuel consumed by the aircraft. The UH-1B, OV-10A, and X-22A are evaluated in this manner in Figure 49.

Conditions 1 and 4 show the change in range-endurance productivity for maximum- and zero-payload conditions with full internal fuel. In both cases, the OV-10A at sea level has approximately twice the range-endurance productivity of the UH-1B, which, in turn, has about three times the capability of the X-22A. The increase of maximum range-endurance productivity that resulted when altitude was increased by

10,000 feet was 41, 60, and 56 percent for the OV-10A, UH-1B, and X-22A, respectively, as indicated by Conditions 4 and 7.

VELOCITY IN LEVEL FLIGHT AND CLIMB

The maximum velocity in level flight is of prime importance for particular V/STOL aircraft where speed is required for interception, surprise attack, escape, or rapid transit. Generally, other considerations such as range, endurance, or payload capability will have equal bearing on the ability of an aircraft to fulfill mission requirements. In most cases, rate of climb is as important as the above-mentioned characteristics for any aircraft, since it is desirable to reach cruise altitude quickly in order to utilize improved thermal and propulsive efficiencies. For high-speed aircraft, optimum climb rates computed from a total energy analysis will also provide significant performance gains.

The maximum velocities of the UH-1B, X-22A, and OV-10A using normal rated power at sea level and at 10,000 feet are compared in Figure 50 for level flight. Whereas the previous comparisons have shown that the X-22A is payload and range-deficient, Figure 50 shows the superior velocity capability of this aircraft. This plot also indicates the velocity limitation of the helicopter at altitude which restricts its envelope of effective operation to relatively low altitudes.

Comparisons of climb velocity with aircraft weight, such as those illustrated in Figure 51, are essential for total evaluation of V/STOL aircraft, since it is important for the aircraft to reach operating altitude quickly, as mentioned above. Figure 51 illustrates the superior climb performance of the X-22A, which is achieved by virtue of its higher power-to-weight ratio that is required for VTOL operation. While the X-22A can reach its operating altitude more quickly than the OV-10A, its fuel requirements in climb for the

higher total weight conditions of the two aircraft are larger than those of the STOL aircraft (Figure 52). At 10,000 feet, the fuel for climb required by the UH-1B helicopter is approximately the same as that for the X-22A, showing the low efficiency of the helicopter at altitude. The superior climb performance of the X-22A is further illustrated by comparisons of the horizontal distance covered by each aircraft while climbing (Figure 53). It should be noted that the curves of Figures 51, 52, and 53 do not represent optimized conditions.

Because of the high power requirements and correspondingly high rates of fuel consumption in the vertical take-off and landing modes of flight, time spent in vertical flight will generally be of short duration for most range and endurance missions of VTOL aircraft. The climb to operating altitude will be largely accomplished by climbing at the rate which will provide best fuel economy, or by flying an optimized velocity-height schedule. For these reasons, the climb characteristics of V/STOL aircraft must be thoroughly analyzed with particular attention given to the fuel required, as shown in Figure 52, or to the calculation of the optimum climb schedule for best range or endurance.

The previous comparisons have illustrated the relative performance limits of the helicopter and the increased performance of the tilt-duct and deflected-slipstream types for an increase of altitude. The helicopter will generally be excluded when consideration is given to the selection of an aircraft for missions requiring high-level operation at relatively high velocity. However, a comparison of service and hover ceilings in Figure 54 shows that the UH-1B is capable of operating at altitudes up to 20,000 feet at minimum design mission weight.

TAKEOFF CHARACTERISTICS

The sacrifice of performance for VTOL capability has been demonstrated in this report for several aircraft of recent design. Until technology has advanced to the point where VTOL flight can be achieved so that the overall performance of VTOL aircraft is nearer that of STOL or conventional aircraft, it is doubtful that any VTOL design will be chosen for military missions in preference to a STOL aircraft when vertical takeoff and landing is not an absolute requirement.

The importance of fuel requirements for the vertical phase of flight has already been emphasized. It is also important that STOL aircraft be thoroughly evaluated to determine STOL capability and that STOL performance of VTOL aircraft be analyzed as well. If the sacrifice of level-flight performance is not too great, a V/STOL aircraft normally operating as a STOL might possibly be selected as the best aircraft for particular missions when only occasional VTOL is required.

Figure 55 illustrates a comparison of the STOL OV-10A and the X-22A operating in the STOL mode. Since many true VTOL aircraft, such as the X-22A, will offer superior STOL capability in terms of takeoff distance required and distance over a 50-foot obstacle, the fuel consumed in takeoff and the payload carried will determine whether the VTOL or STOL aircraft will be selected to fulfill mission takeoff requirements.

SUMMARY OF PHASE 2 EVALUATIONS

The preceding evaluation and comparison of the OV-10A, UH-1B, and X-22A aircraft were performed to illustrate methods of comparison that could be used to evaluate the performance and capability of V/STOL aircraft. In some instances which depend upon mission requirements, it may be convenient or necessary to devise other parameters or to present data in

other forms to adequately express the performance of V/STOL aircraft. This will depend upon particular mission requirements and the degree of emphasis assigned to particular capabilities, such as range and endurance, for the mission being considered. It was not possible or within the scope of this study to thoroughly examine the takeoff, landing, and climb-out phases of flight and to relate the performance characteristics of V/STOL aircraft in those flight regimes to level-flight performance except in a cursory manner. Further study is needed for those phases of flight in addition to a need for further analysis of altitude effects on V/STOL performance.

While this analysis was performed for aircraft in the clean or low-drag configuration, the same methods may be used for actual mission configurations when specific missions are considered. At a later date, when more actual flight-test information becomes available on the performance of V/STOL aircraft in vertical and transitional flight, a more thorough analysis of the total performance of V/STOL aircraft will be possible. More information is required that is relative to fuel flow requirements and velocities in vertical and transitional flight for a larger number of STOL and V/STOL configurations.

All of the evaluation methods presented herein that are based on fuel flow rate or equivalent thermal power are easily applicable to any type of aircraft. An attempt has been made to show that total evaluation of V/STOL aircraft may be performed using fuel consumption rates only. A comparison of performance on this basis appears to be especially applicable to V/STOL aircraft which are powered by turbojet or turbine engines because of the large changes in performance with varying velocity and altitude that have been shown to be strongly related to thermal efficiency.

A summary of important level-flight performance parameters that are useful in the evaluation of V/STOL aircraft for military missions is given in Table IV.

TABLE IV

IMPORTANT LEVEL-FLIGHT PARAMETERS FOR EVALUATING MISSION PERFORMANCE REQUIREMENTS

Primary Mission Capabilities	Type Mission	Type Aircraft	Important Parameters	Units
Range, Payload	Intra-Theater Operations	Cargo or Troop Transport	W_u	Pounds of payload
			V/Q	Nautical miles per pound of fuel
			W_fV/Q	Nautical miles
			W_uQ/W_f	Pounds of payload per hour
			W_uV/Q	Pounds of payload per pound of fuel times nautical miles
Payload, Speed	Short-Range Intercep-tion	Fighter	W_u	Pounds of payload
			V	Nautical miles per hour
			W_uV	Pounds of payload times nautical miles per hour
	Close Sup-port or Assault	Armed Cargo or Troop Transport	W_uQ/W_f	Pounds of payload per hour
Range, Endurance	Long-Range Surveil-lance	Reconnais-sance	V/Q	Nautical miles per pound of fuel
			W_f/Q	Hours
			W_fV/Q	Nautical miles
			W_vV/Q^2	Nautical miles per pound of fuel times hours
Endurance	Short-Range Observa-tion	Reconnais-sance	W_f/Q	Hours
Payload, Endurance	Armed Reconnais-sance	Reconnais-sance	W_u	Pounds of payload
			W_u/Q	Pounds of payload per pound of fuel times hours
			W_f/Q	Hours
	Short-Range Heavy-Lift Operations	Crane	W_uQ/W_f	Pounds of payload per hour
Range, Speed	Long-Range Strike or Escort	Fighter	V	Nautical miles per hour
			V/Q	Nautical miles per pound of fuel
			W_fV/Q	Nautical miles
	Rescue	Utility		

PHASE 3. INVESTIGATION OF NONDIMENSIONAL PARAMETERS FOR IDENTIFICATION OF FLOW REGIMES FOR V/STOL AIRCRAFT

The Reynolds number has long been used in aerodynamics to identify regimes of flow for correlation of aerodynamic forces which act on bodies due to the relative motion between the bodies and the fluid mass. This parameter is especially useful in predictions of the forces which will act on a full-scale body from tests of a smaller model. Comparisons of body forces may also be made from data obtained under different test conditions, as long as the comparisons are made at a given Reynolds number. The use of nondimensional parameters such as the Reynolds number also provides a basis for comparison of V/STOL aircraft when the limitations of the use of these "numbers" is observed.

LIMITATIONS OF REYNOLDS NUMBER

The use of Reynolds number is valid for comparison of force coefficients only when the bodies being compared are geometrically similar. Thus, the Reynolds number may be used in the prediction of lift and drag of a full-scale aircraft from the forces acting on a model of the same aircraft. It may also be used to correlate the forces acting on two configurations of the same aircraft or model, provided no geometric changes are involved.

Sometimes the Reynolds number is used in the comparison of lift and drag characteristics of different full-scale aircraft of a particular type, considering the vehicles to be essentially geometrically similar. For fixed-wing conventional airplanes or helicopters, the results are valid only in a general sense, since the validity of such comparisons depends upon the degree of geometric similarity that exists among the various vehicles. Certainly, the Reynolds number cannot be used with any degree of reliability for the

comparison of a group of V/STOL aircraft having widely different geometric configurations.

The Reynolds number also fails to account for lift and drag forces which are the result of the expenditure of power for increased circulation, since this parameter is related only to aerodynamic characteristics by the variables ρ, V, and μ. This is particularly significant for V/STOL aircraft which derive their high-lift capability by utilizing waste energy or by the expenditure of power for high-lift systems of some type.

DIMENSIONLESS PARAMETERS BASED ON FUEL FLOW RATE

In view of the above restrictions, new parameters are needed which could be used in a manner similar to the use of Reynolds number, but which would include a variable to account for power effects on lift and drag. Toms (Reference 58) has suggested the use of fuel flow rate Q as a new variable which would be applicable to any type of aircraft and which would account for power augmentation. Toms has derived a new number, $Q/\rho V^3 \ell$, which would be usable for widely different systems, be readily evaluated, and which would not depend upon specific system properties. This new number, called the Toms number, should indeed be valuable as a basis for comparing the force coefficients of aircraft, but only for geometrically similar conditions.

Since Toms number includes the variable "ℓ", geometric restrictions on the use of Reynolds number apply as well to the use of this new parameter. It would appear that new parameters which use the fuel flow rate Q as a variable may be used to identify flow regimes for V/STOL aircraft having lift or thrust augmentation in the same manner that Reynolds number is used with aircraft which do not use power-augmented systems. The use of such parameters would be restricted to the following cases:

A. Correlation of the force coefficients of a full-scale aircraft and those of a smaller scale model.

B. Comparison of the force coefficients of a single aircraft with and without power-augmented circulation.

C. Comparison of different aircraft or models having the same general geometric configuration.

D. Comparison of aircraft or models having different geometric configurations when the geometric variable included in the dimensionless number can be chosen such that the differences of configuration can be accounted for.

Condition C refers to comparison of aircraft of the same type, such as single-rotor helicopters or tilt-wing vehicles. Condition D refers to another general type of comparison where the geometric variable can be chosen for each aircraft to account for difference in size or manner of lift or thrust generation. For instance, it might be possible to compare the lift coefficients of a helicopter to those of a fixed-wing vehicle by using disc area and wing planform area as variables for the identification of similar flow regimes for the two vehicles. Experimental data are required, however, for verification of the use of dimensionless parameters in this manner.

From the variables ρ, μ, V, ℓ, and Q, several new dimensionless numbers may be derived which can be used to identify flow regimes for comparison of forces acting on geometrically similar bodies. The Reynolds number and Toms number are two of these. By multiplying Toms number $Q/\rho V^3\ell$ by the inverse of Reynolds number $\mu/\rho V\ell$, still another number can be obtained which includes the dynamic viscosity μ. This number is $\mu Q/\rho^2 V^4 \ell^2$. The inclusion of μ allows comparison of force

coefficients on bodies tested in fluids of different dynamic viscosity, or it allows for changes of dynamic viscosity in the test fluid. When aircraft are evaluated in the air, this new number will account for changes in dynamic viscosity which may occur due to temperature variation.

In the case of V/STOL aircraft which operate at relatively high subsonic speeds, compressibility effects may be considered by the use of dimensionless numbers which include the speed of sound as a variable such as $Qa/\rho V^4 \ell$ or $\mu Qa/\rho^2 V^5 \ell^2$.

Unfortunately, a lack of sufficient data for aircraft with and without power-augmented lift systems has prevented a thorough investigation and confirmation of the validity of using the above-mentioned parameters for comparison of V/STOL configurations. Some insight into the use of such numbers may be gained, however, from Figures 56 through 60.

Figures 56 through 60 show plots of lift and drag coefficient for the OV-10A deflected-slipstream-type aircraft in level flight. Values of C_L and C_D were obtained from Reference 8. The curves of Figure 56, which are plotted against Reynolds number, may be compared to those of Figures 57 and 58, which are shown as functions of parameters that include the fuel flow rate Q.

The drag coefficient curves of Figure 56 show fair agreement between sea level and 10,000 feet for the clean aircraft, but the use of Toms number in Figure 57 indicates a greater difference between the lift and drag coefficients at the lower airspeeds. The failure of Toms number to show a closer agreement between the curves for sea level and 10,000 feet than did the Reynolds number may perhaps be related to changes of thermal efficiency with altitude. When the variable μ is introduced as in Figure 58, agreement between the curves is improved. Unfortunately, the data used in these illustrations were of a semiempirical nature which prevented

an analysis to determine the reasons for differences in the agreement or disagreement between the curves when plotted as functions of the various parameters shown. Figure 58 indicates, however, that the parameter $\mu Q/\rho^2 V^4 \ell^2$ may perhaps be more useful than Toms number to establish the flow regimes of V/STOL aircraft.

Some idea of the effects of thermal efficiency change with altitude on the parameters that have the fuel flow rate Q as a variable may be obtained from comparisons of Figures 57 and 58 with Figures 59 and 60, respectively. When Toms number is replaced by the parameter $SHP/\rho V^3 \ell^2$ in Figure 59, the lift and drag coefficient curves for sea level and altitude conditions are almost identical for the clean aircraft. Closer agreement is also obtained in Figure 60 when shaft horsepower is used instead of the fuel flow rate Q. These results indicate that shaft horsepower required should perhaps be selected as the variable to relate lift and drag forces to the power expended for lift or thrust augmentation instead of the fuel flow rate Q. It should be noted, however, that drag values of the armed reconnaissance version of the OV-10A appear to be less than those of the clean version when plotted against the higher values of the parameters using shaft horsepower as a variable. This shows that these parameters may be of doubtful value and that the fuel flow rate Q may be the more significant variable.

SUMMARY OF PHASE 3 INVESTIGATION

Comparisons of the lift and drag coefficients of the OV-10A, for flow regimes defined by dimensionless parameters based on fuel flow rate and shaft horsepower required, indicate that these new parameters may be of significant value for evaluation of V/STOL aircraft. Additional experimental data are required, however, to establish more firmly the relationships of lift and drag forces to these new quantities.

Dimensionless parameters based on fuel flow rate should be especially convenient and useful for evaluation of aircraft having lift and thrust augmentation systems, and for comparisons of the lift and drag characteristics of geometrically similar types of aircraft.

CONCLUSIONS

1. Thermal energy provides a common basis for the evaluation and comparison of V/STOL aircraft performance. The use of thermal energy for this purpose has the advantages of being valid for all aircraft, regardless of geometric configuration or type of propulsion system, and of being easily determined from measurements of fuel flow rate. The method also permits comparison to be made of V/STOL performance on the basis of both propulsive and thermal efficiency. This allows a valid comparison to be made of aircraft having propulsion systems with significantly different thermal efficiencies. For aircraft having lift or thrust augmentation systems deriving energy from auxiliary sources, the energy required for these systems may readily be combined with the thermal energy of the fuel required to obtain equivalent thermal energy as the basis for performance evaluation.

2. Comparisons of a number of jet- or turbine-powered V/STOL aircraft have shown that the relative range and endurance capability of each aircraft was generally consistent when comparisons were made on the basis of both equivalent thermal energy and shaft horsepower. This result occurred since variations of thermal efficiency were smaller than variations of the propulsive efficiencies of the aircraft being considered. It is concluded that the overall performance capability of V/STOL aircraft is more strongly related to propulsive efficiency than to thermal efficiency, but only when the power plants of the aircraft being compared are of the same general type.

3. Methods based on total mechanical energy (kinetic plus potential) should be utilized to optimize climbing performance of high-speed V/STOL aircraft. The increased

performance gained by optimization of climb schedules is heavily dependent on aircraft maneuverability, pilot technique, climb speed, altitude, and takeoff requirements. For these reasons, the decision to optimize performance will depend on the designated operational speed and altitude requirements, the degree of accuracy desired, and other factors as mentioned above. Calculations for a typical V/STOL aircraft indicate that, in general, optimization of climb schedules using total energy methods will be profitable when cruising speed and altitude exceed 250 knots and 30,000 feet, respectively.

RECOMMENDATIONS

The greatest advantage of using equivalent thermal energy for performance evaluation of aircraft is that comparisons of aircraft of entirely different design may be made on the basis of total aircraft efficiency. Thus, the method includes the thermal efficiency of the power plant, which is important in terms of fuel economy, range, and endurance. For this reason, thermal energy should be utilized as the basis for comparisons of the performance of two or more aircraft, especially when these aircraft are powered by different types of engines. In cases where the aircraft are powered by similar types of power plants, comparisons may be made considering propulsive efficiency alone, since differences in thermal efficiency will generally be small. However, since equivalent thermal energy is easily determined and since comparisons based on equivalent thermal energy will provide a more accurate analysis, it is recommended that this method be adopted in all cases.

It is also suggested that the decision to optimize performance of V/STOL aircraft by the use of total mechanical energy methods could be based on preliminary calculations to determine the ideal performance gain that might be achieved. This could be accomplished using estimated performance data or flight-test data if available. If the performance gain appeared to be significant, a flight-test program could then be initiated to acquire the data necessary to accurately establish the optimum climb conditions of the aircraft for designated mission cruise speeds and altitudes. The actual performance gains would then be found from further flight tests of the aircraft, during which the ability of both aircraft and pilot to meet the requirements of the optimized climb schedule would be established.

Additional investigation to acquire experimental data is needed to confirm the significance of using nondimensional parameters based on fuel flow rates or power requirements to define flow regimes for V/STOL aircraft. There is also a need for standardization of data presentation methods, flight-test procedures, and notation in the V/STOL field to simplify correlation of published data. Additional work is also required to examine more thoroughly the use of total thermal and mechanical energy techniques and to extend them to higher speed jet V/STOL aircraft as sufficient data become available.

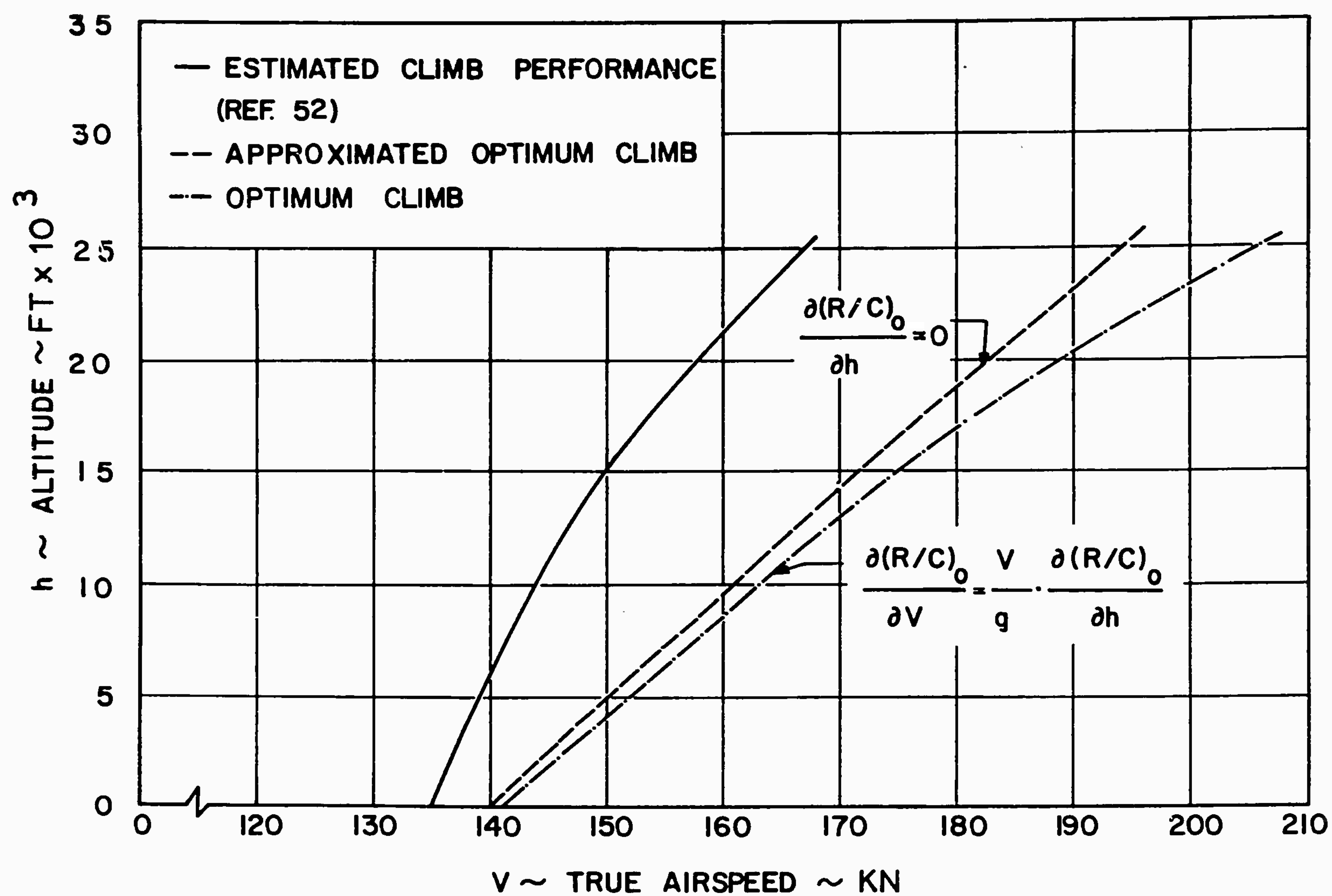

Figure 1. Comparison of Optimum Climb Schedules With a Typical Non-Optimized Climb Performance Curve for the X-22A Aircraft - W_g = 15,930 Pounds, Normal Rated Power.

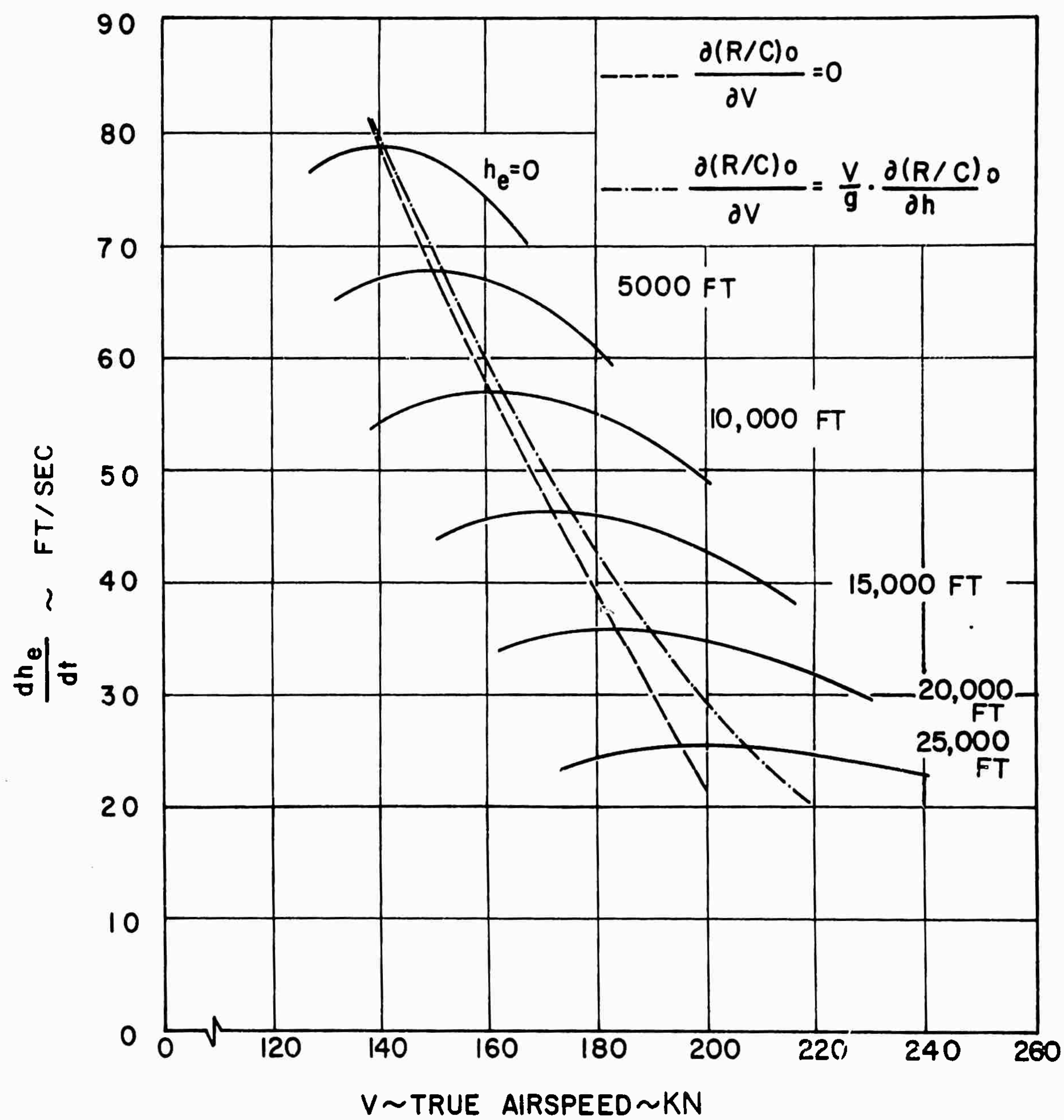

Figure 2. Variation of the Energy Height Time Derivative as a Function of True Horizontal Airspeed - X-22A, W_g = 15,930 Pounds, Climbing Flight at Normal Rated Power.

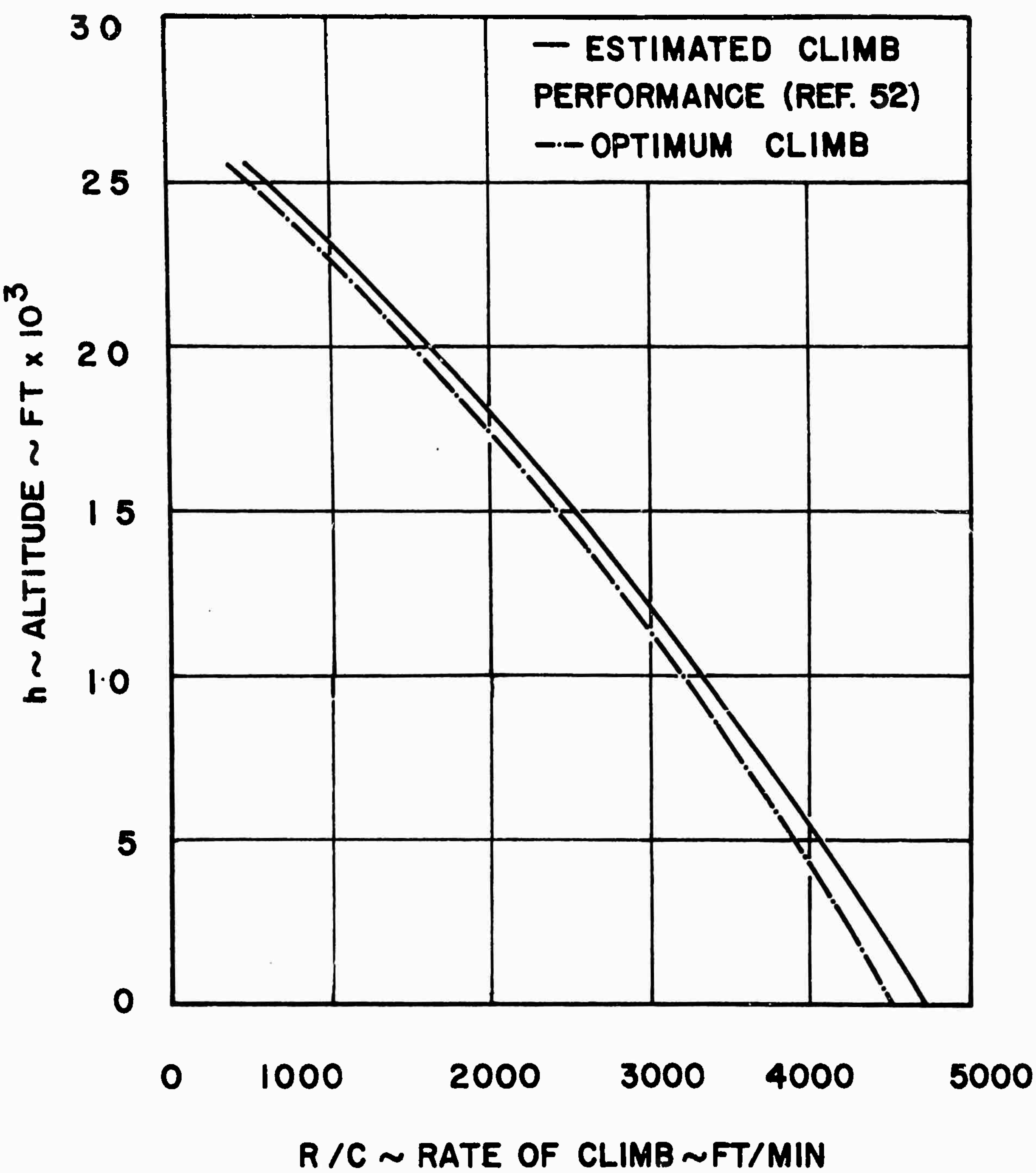

Figure 3. Effect on Rate of Climb of the X-22A Due to Optimization of the Climb Schedule - W_g = 15,930 Pounds, Normal Rated Power.

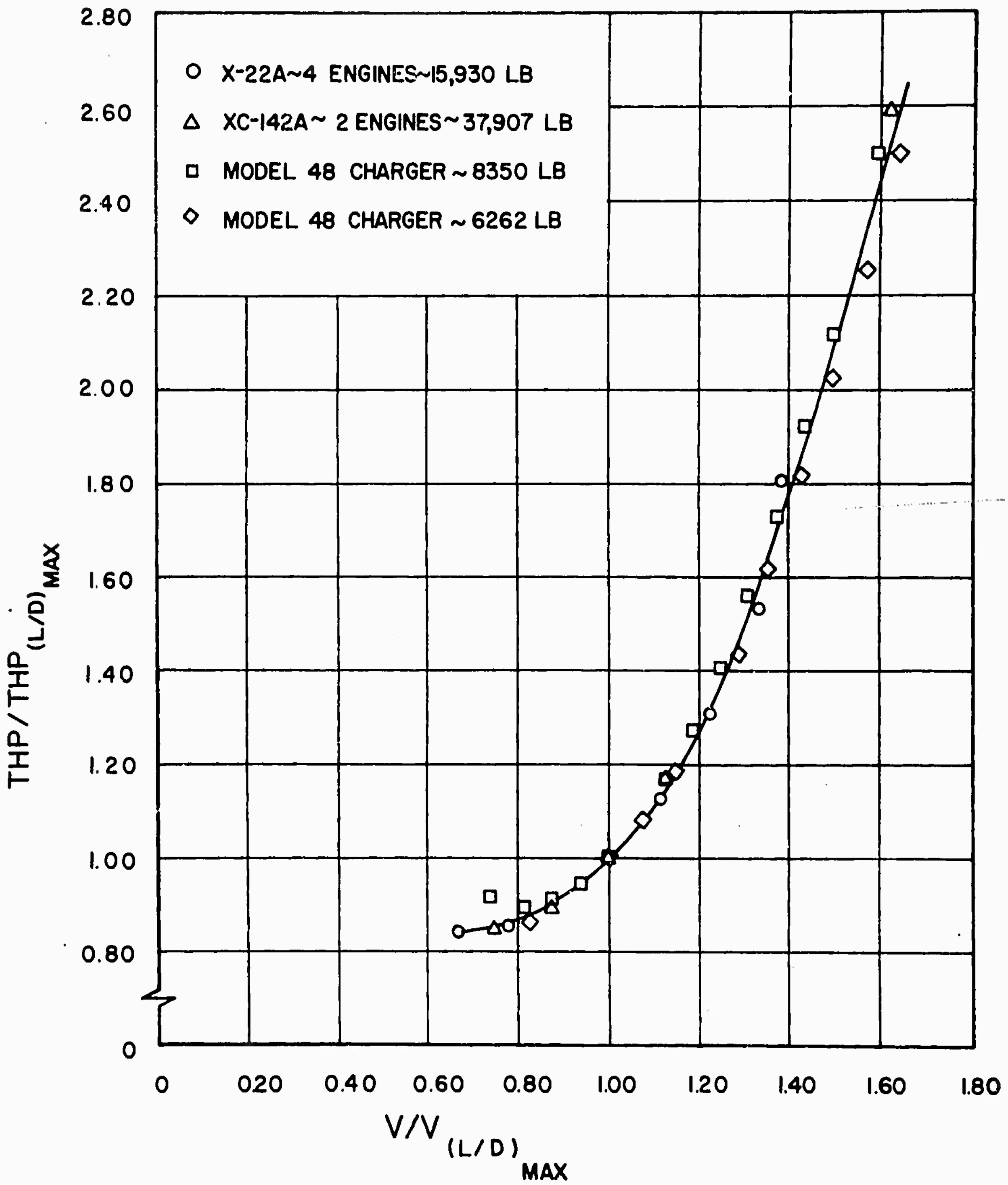

Figure 4. Generalized Thrust Horsepower Required Data for the X-22A, Model 48 Charger, and XC-142A Aircraft - Level Flight, Standard Day, Normal Rated Power.

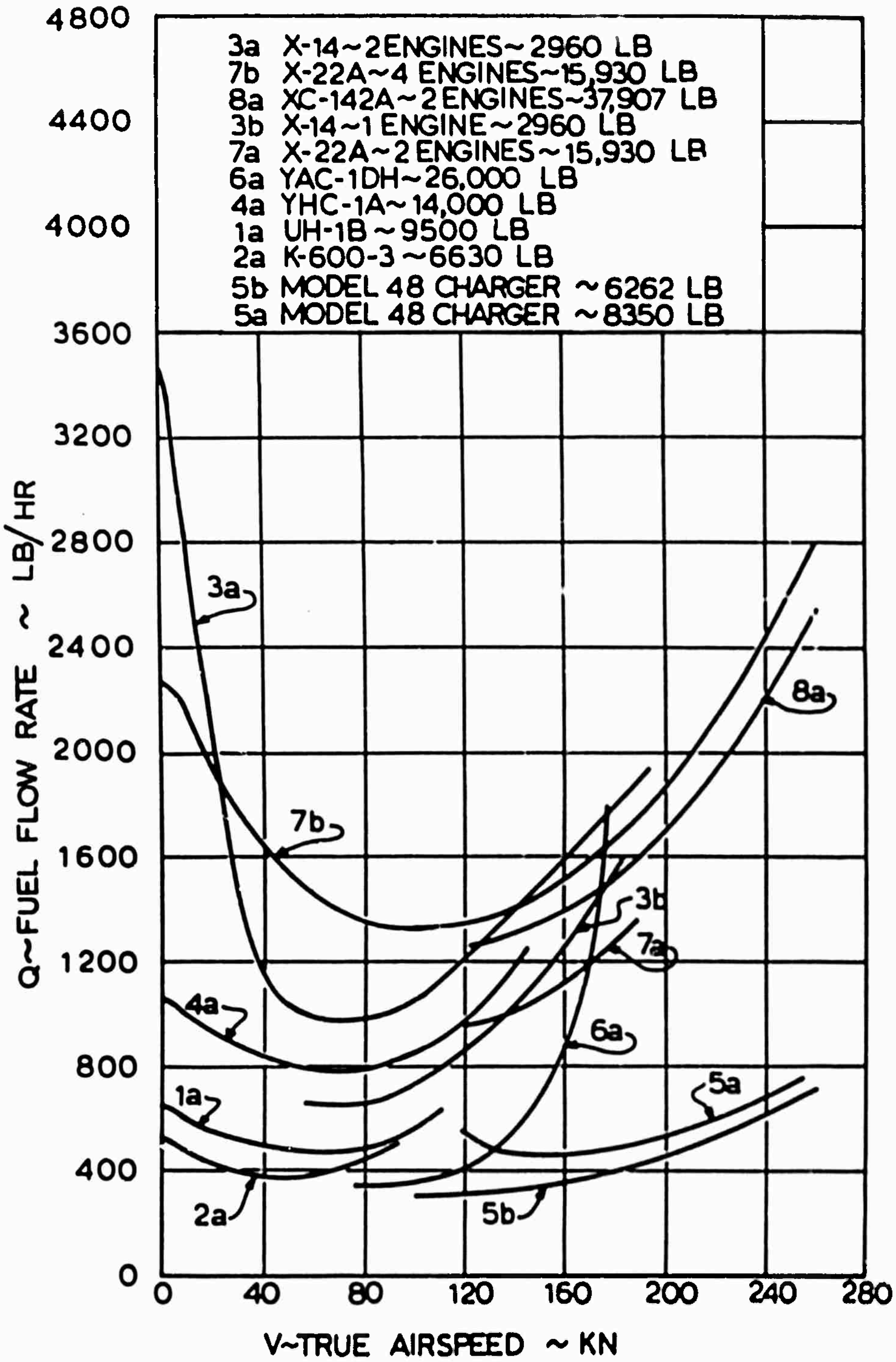

Figure 5. Effect of True Airspeed on the Fuel Flow Rate of Various Aircraft - Sea Level, Standard Day, Level Flight, Normal Rated Power.

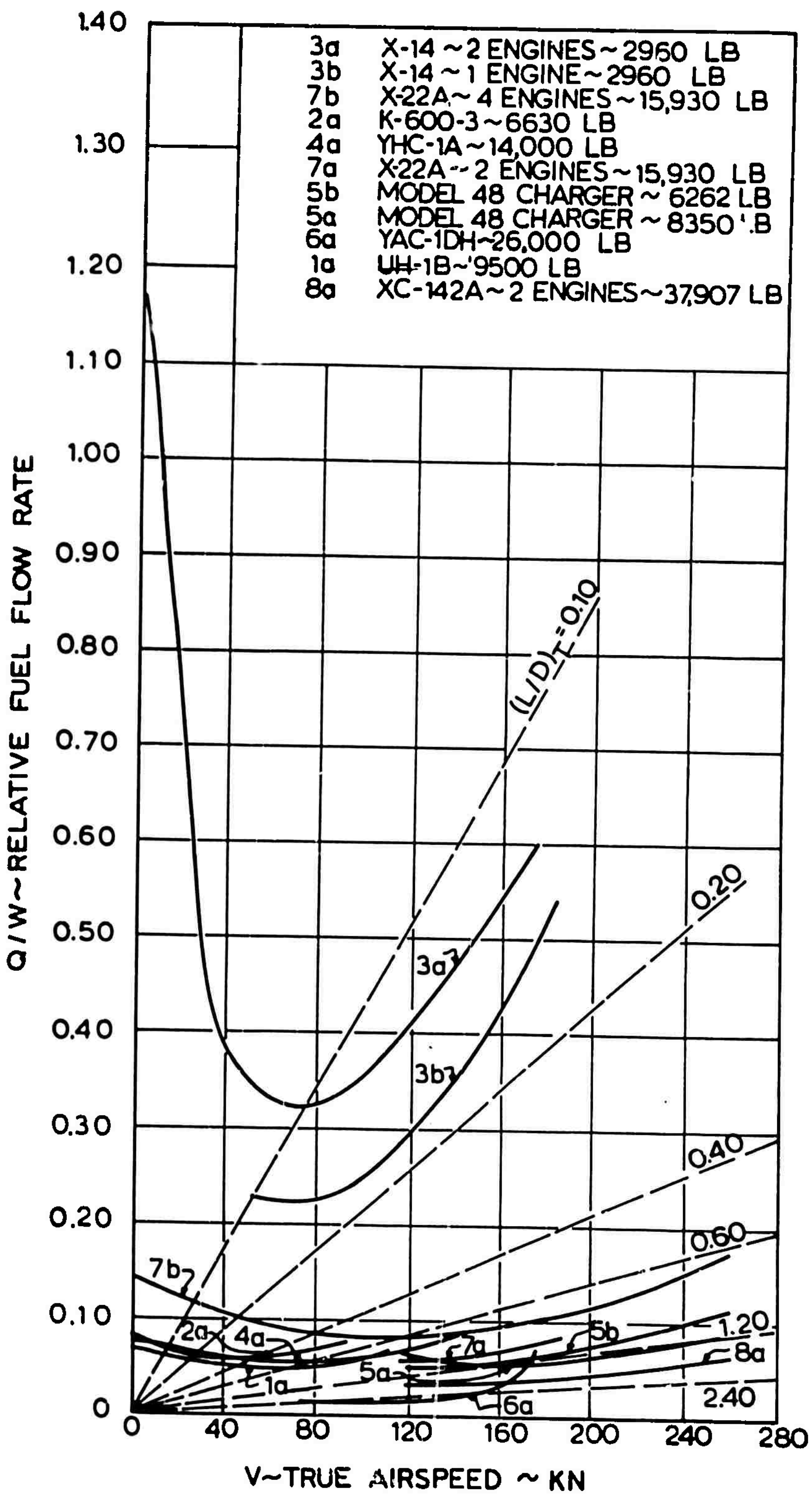

Figure 6. Effect of True Airspeed on the Relative Fuel Flow Rate of Various Aircraft - Sea Level, Standard Day, Level Flight, Normal Rated Power.

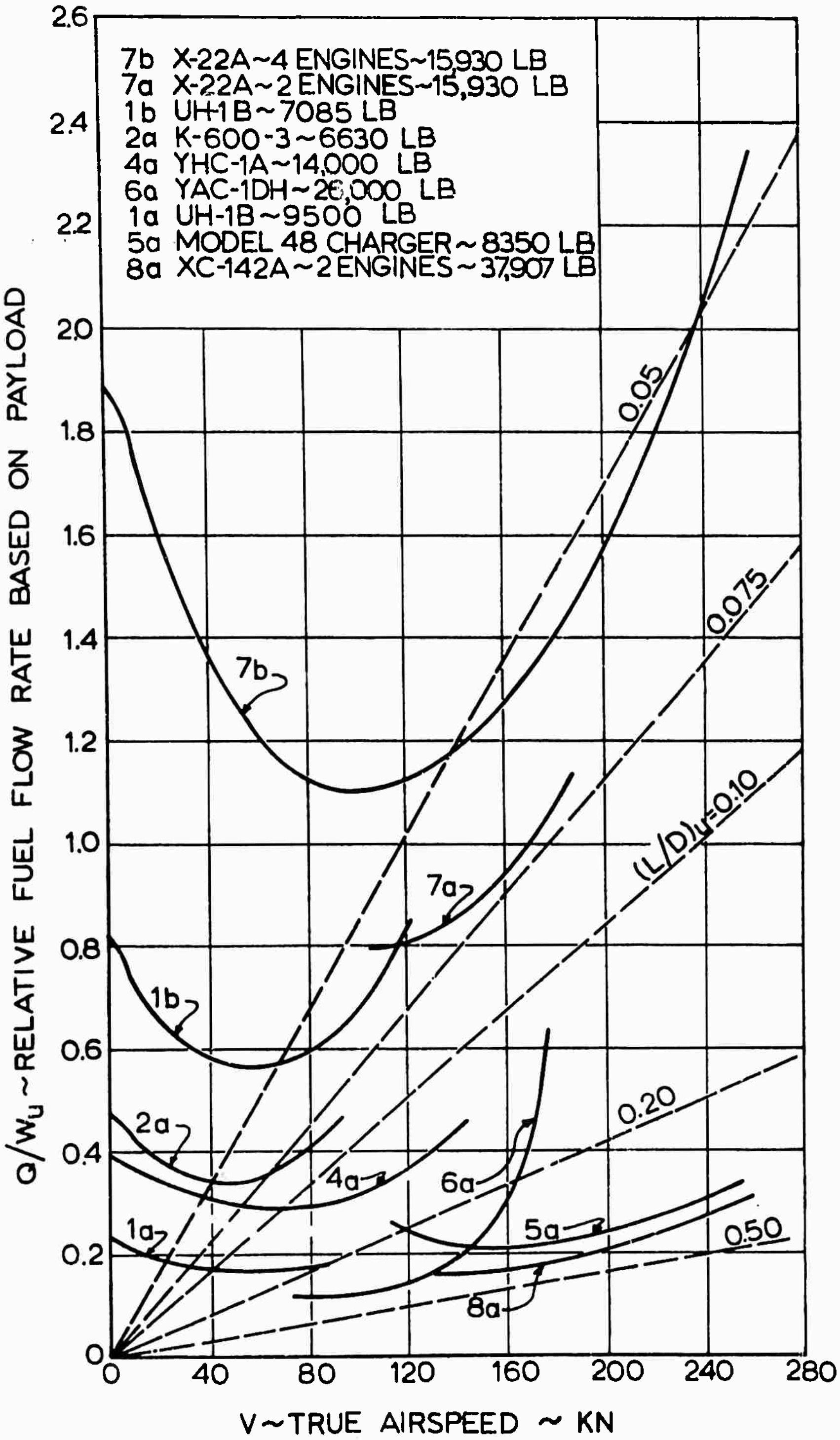

Figure 7. Comparisons of Relative Fuel Flow Rate Based on Payload for Various Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

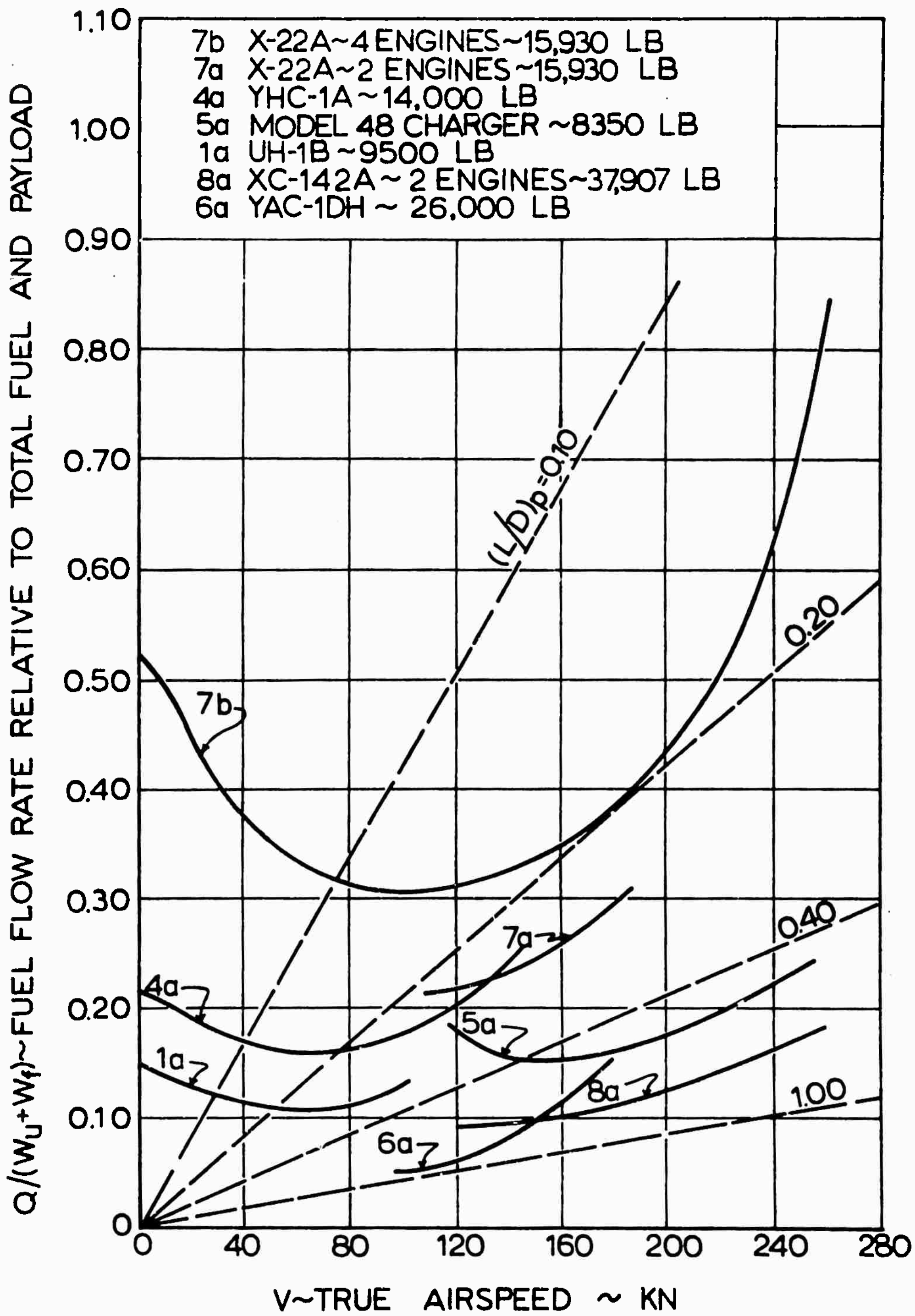

Figure 8. Comparisons of Relative Fuel Flow Rate Based on Fuel Load Plus Payload for Various Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

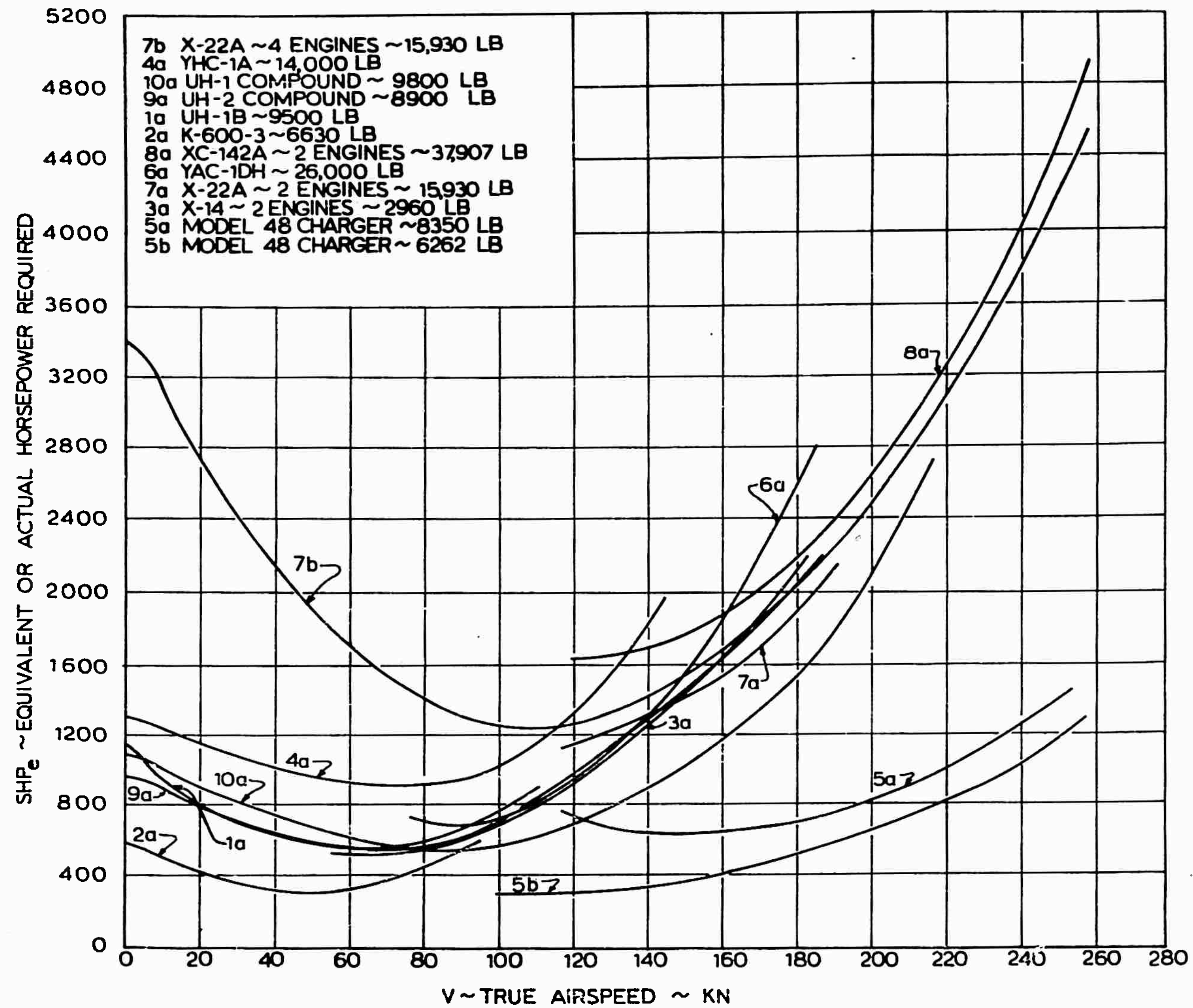

Figure 9. Equivalent Shaft Horsepower Required by Various Aircraft in Level Flight at True Airspeed - Sea Level, Standard Day, Normal Rated Power.

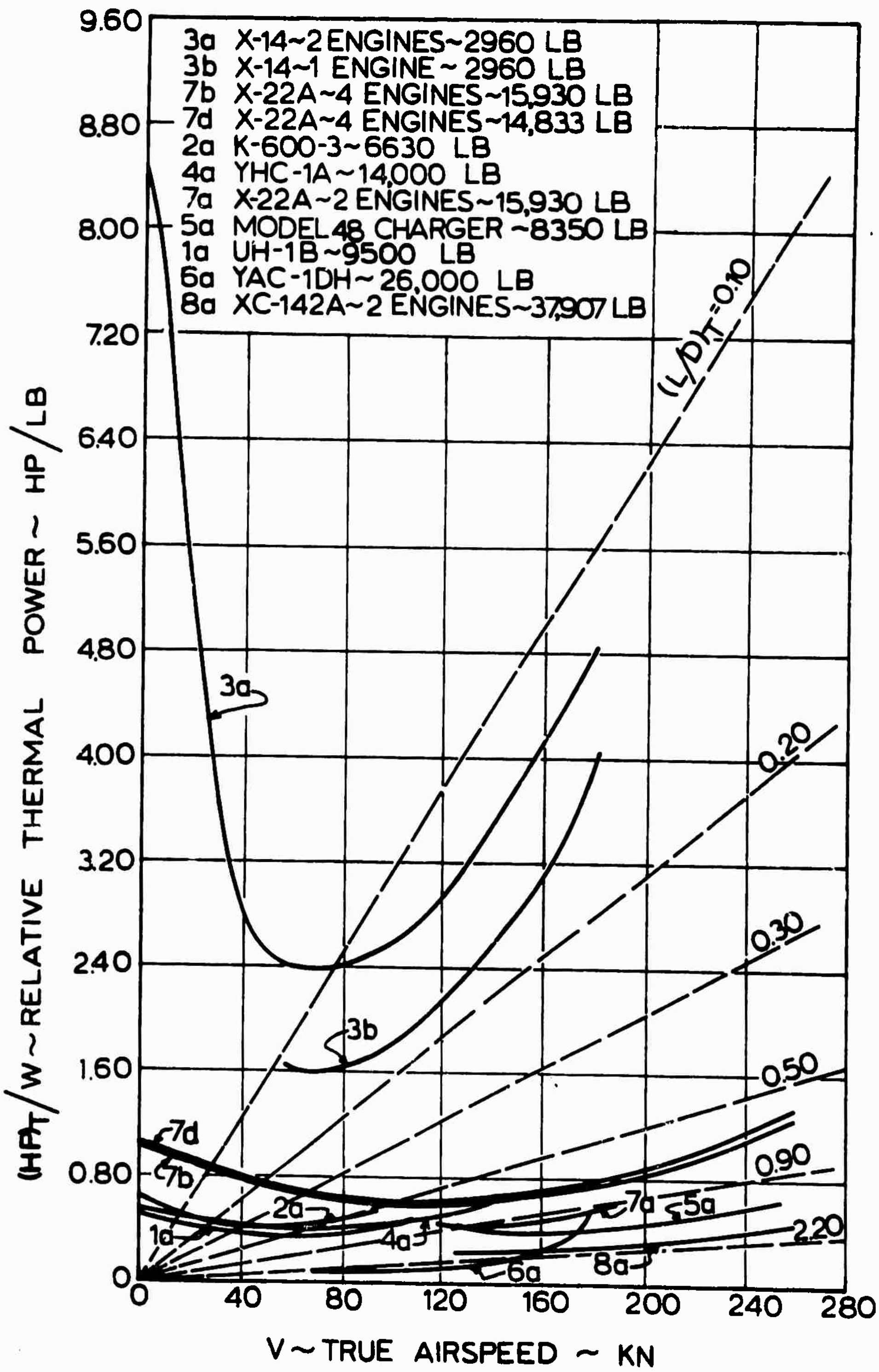

Figure 10. The Effect of True Airspeed on the Relative Thermal Power Required by Various Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

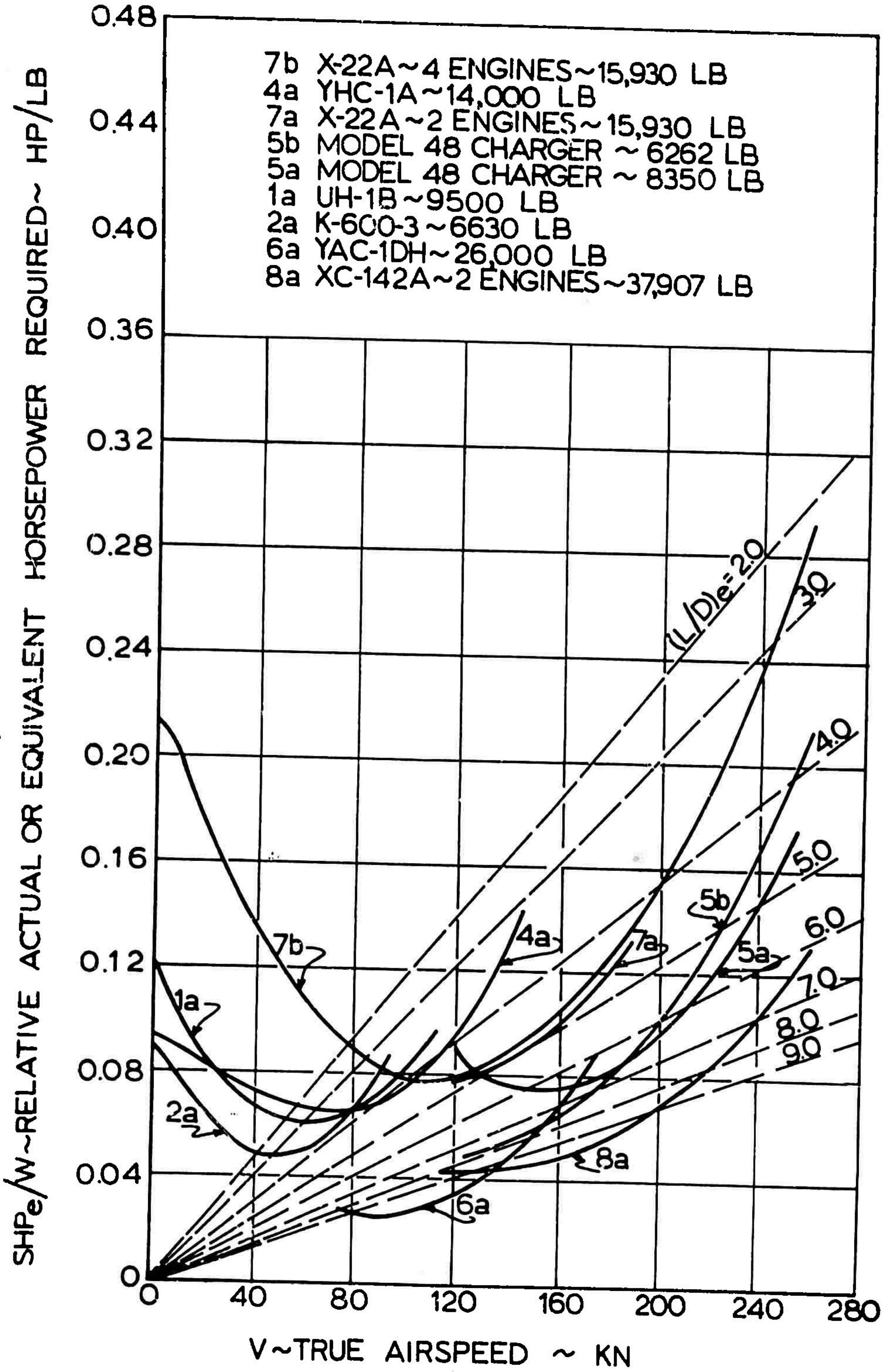

Figure 11. The Effect of True Airspeed on the Relative Actual or Equivalent Power Requirements of Various Aircraft - Sea Level, Standard Day, Level Flight, Normal Rated Power.

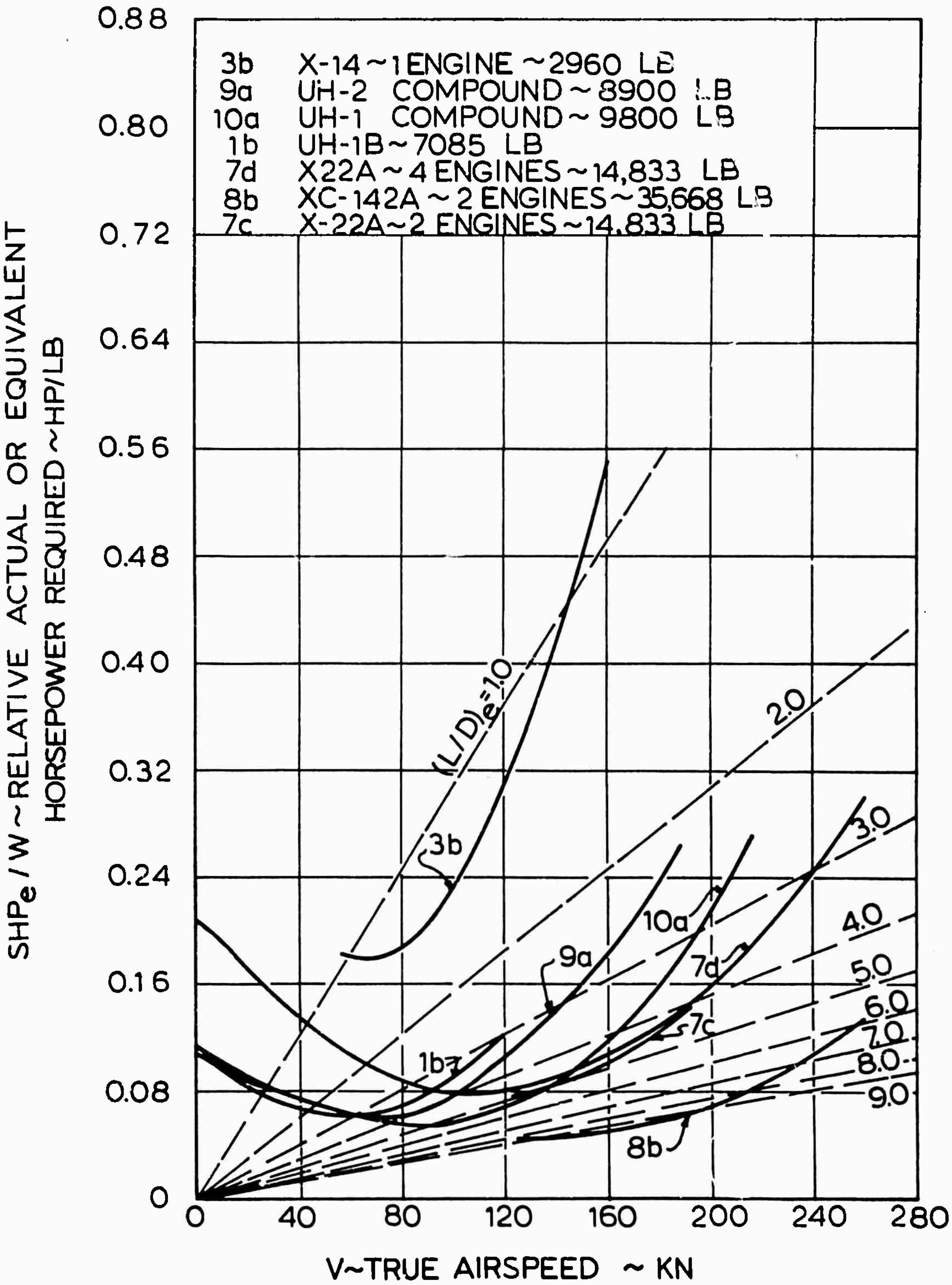

Figure 12. The Effect of True Airspeed on the Relative Actual or Equivalent Power Requirements of Various Aircraft - Sea Level, Standard Day, Level Flight, Normal Rated Power.

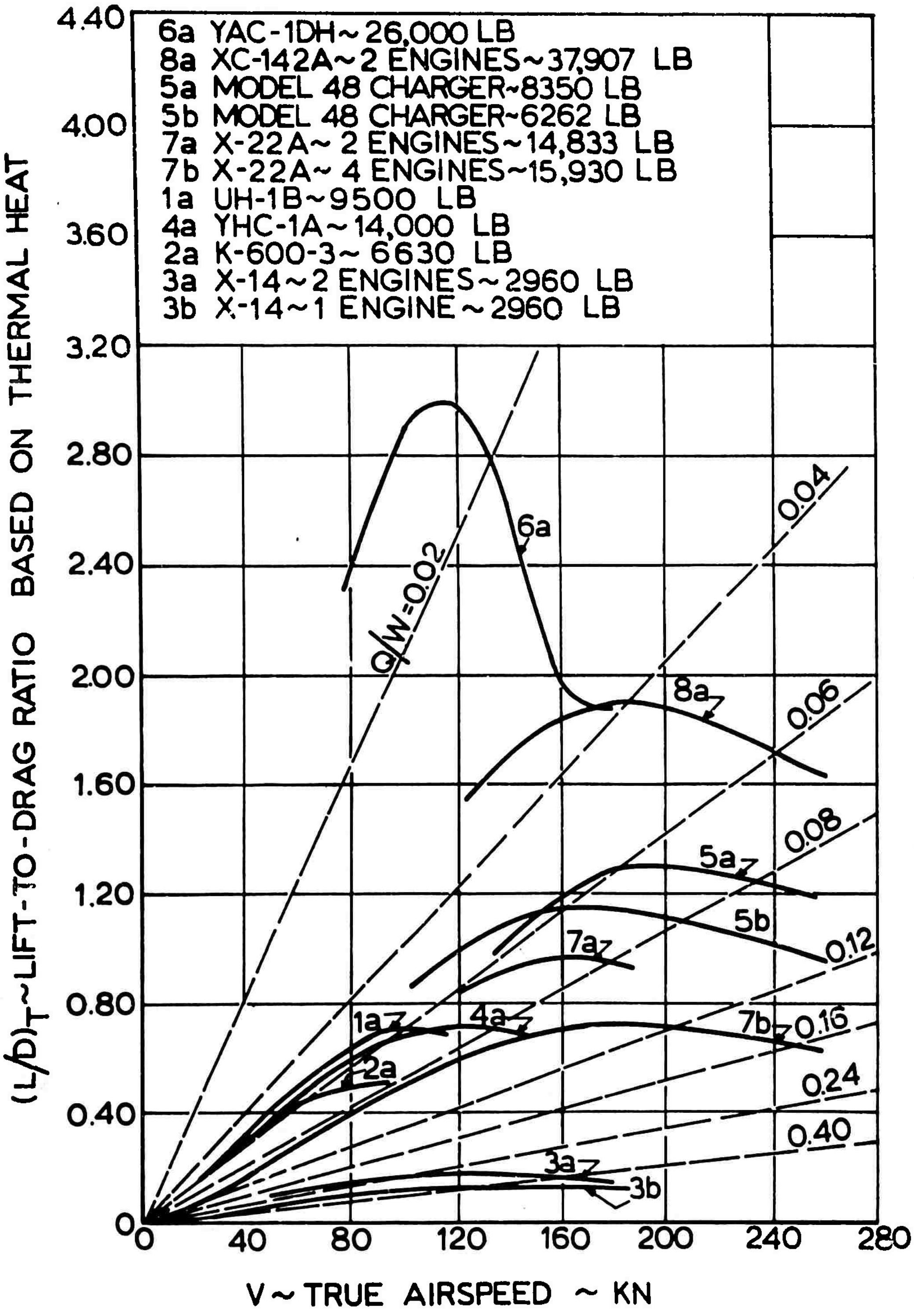

Figure 13. A Comparison of Lift-to-Drag Ratio Based on Thermal Heat for Various Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

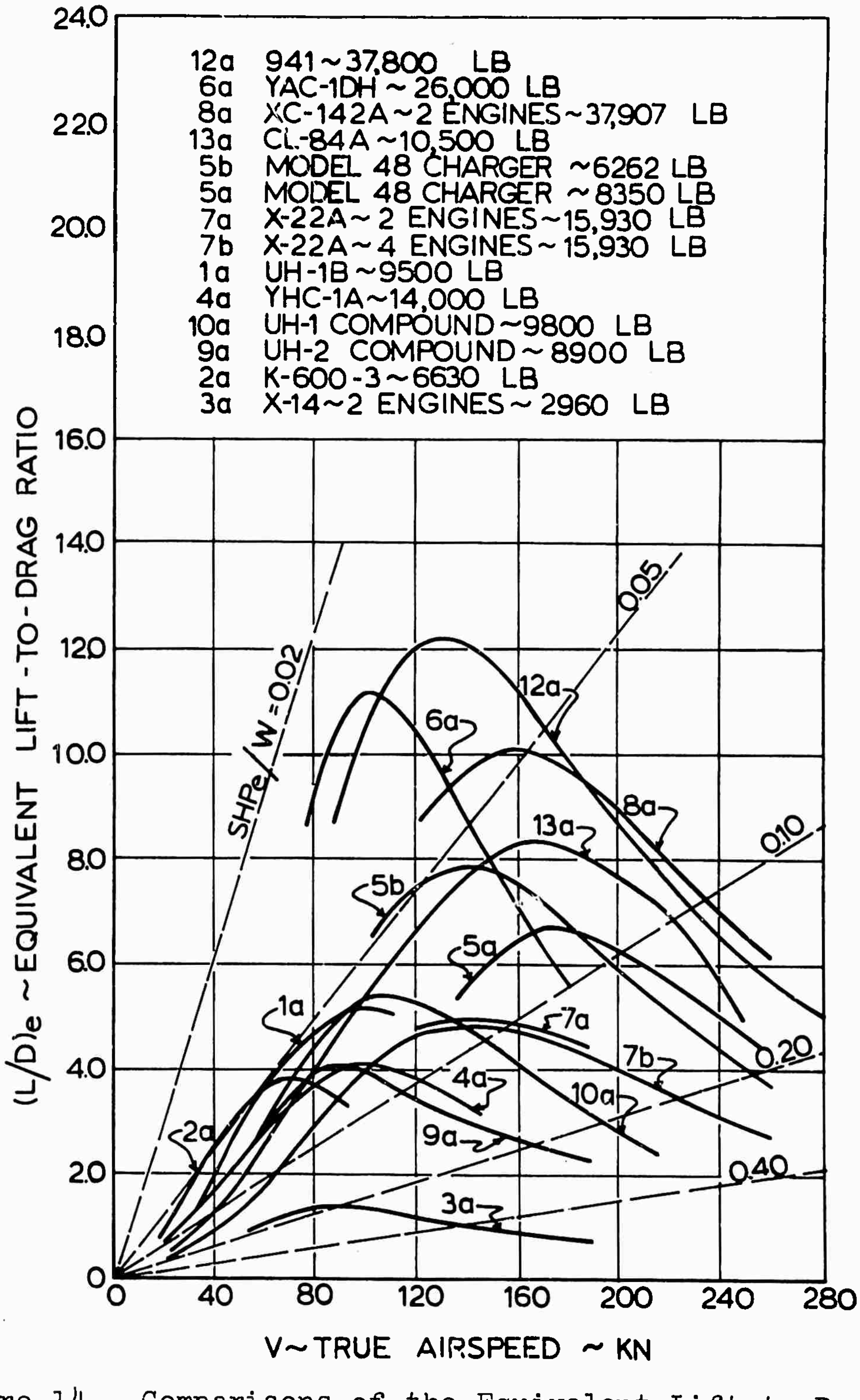

Figure 14. Comparisons of the Equivalent Lift-to-Drag Ratio for Various Aircraft in Level Flight - Sea Level, Standard Day.

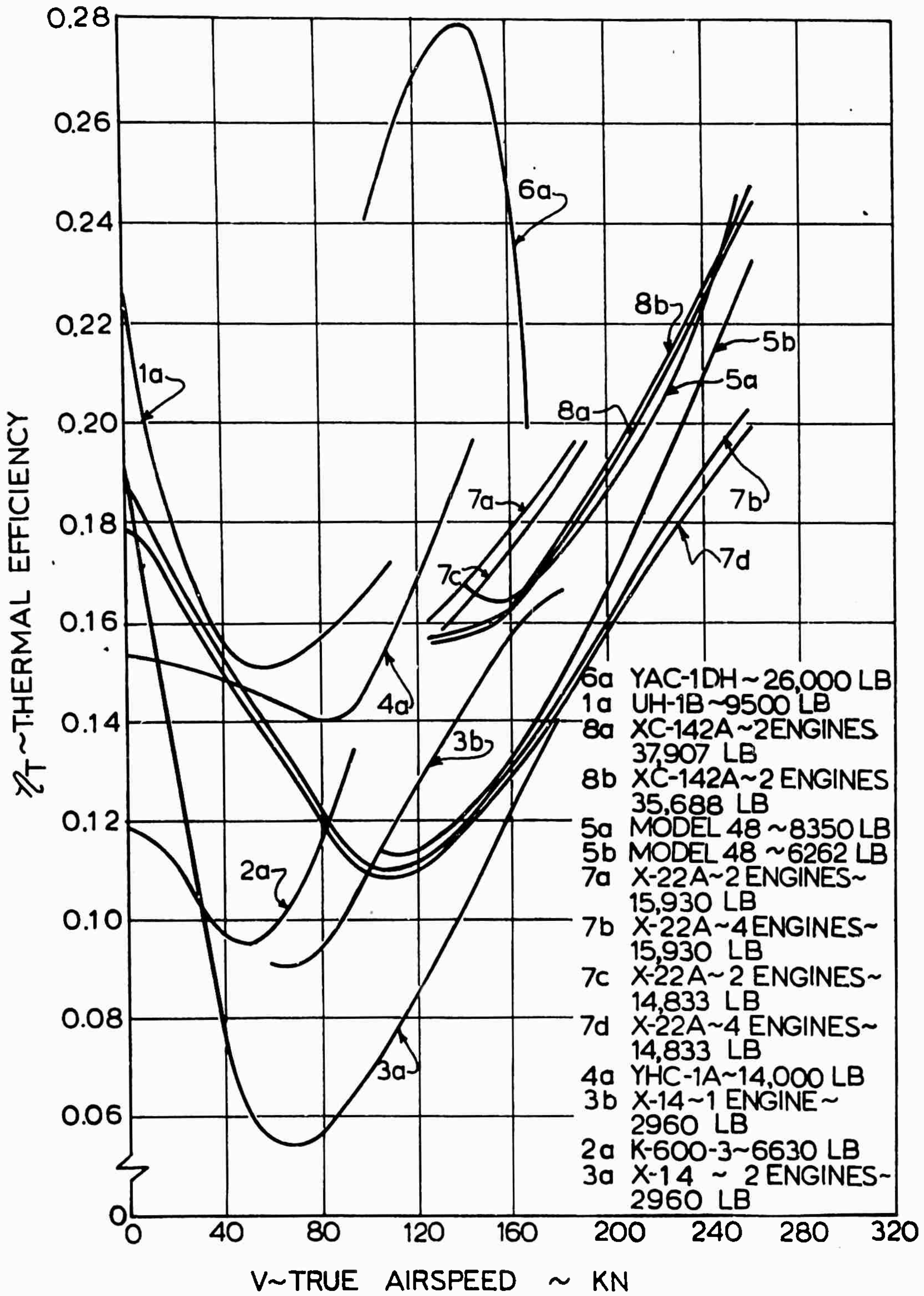

Figure 15. The Effect of Airspeed on the Thermal Efficiency of Different Types of Aircraft for Various Gross Weights and Engine Operating Conditions - Sea Level, Standard Day, Level Flight, Normal Rated Power.

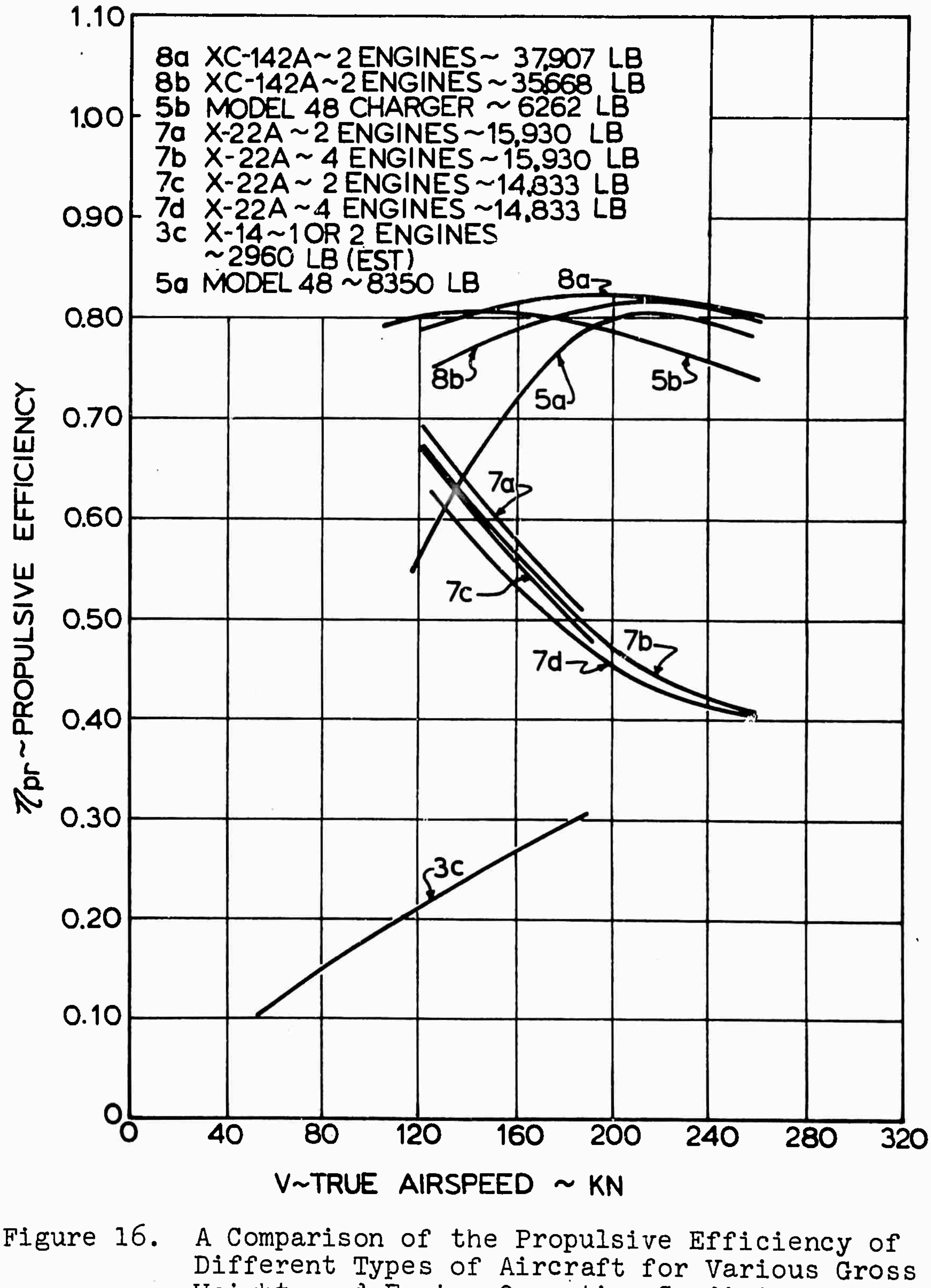

Figure 16. A Comparison of the Propulsive Efficiency of Different Types of Aircraft for Various Gross Weights and Engine Operating Conditions - Sea Level, Standard Day, Level Flight, Normal Rated Power.

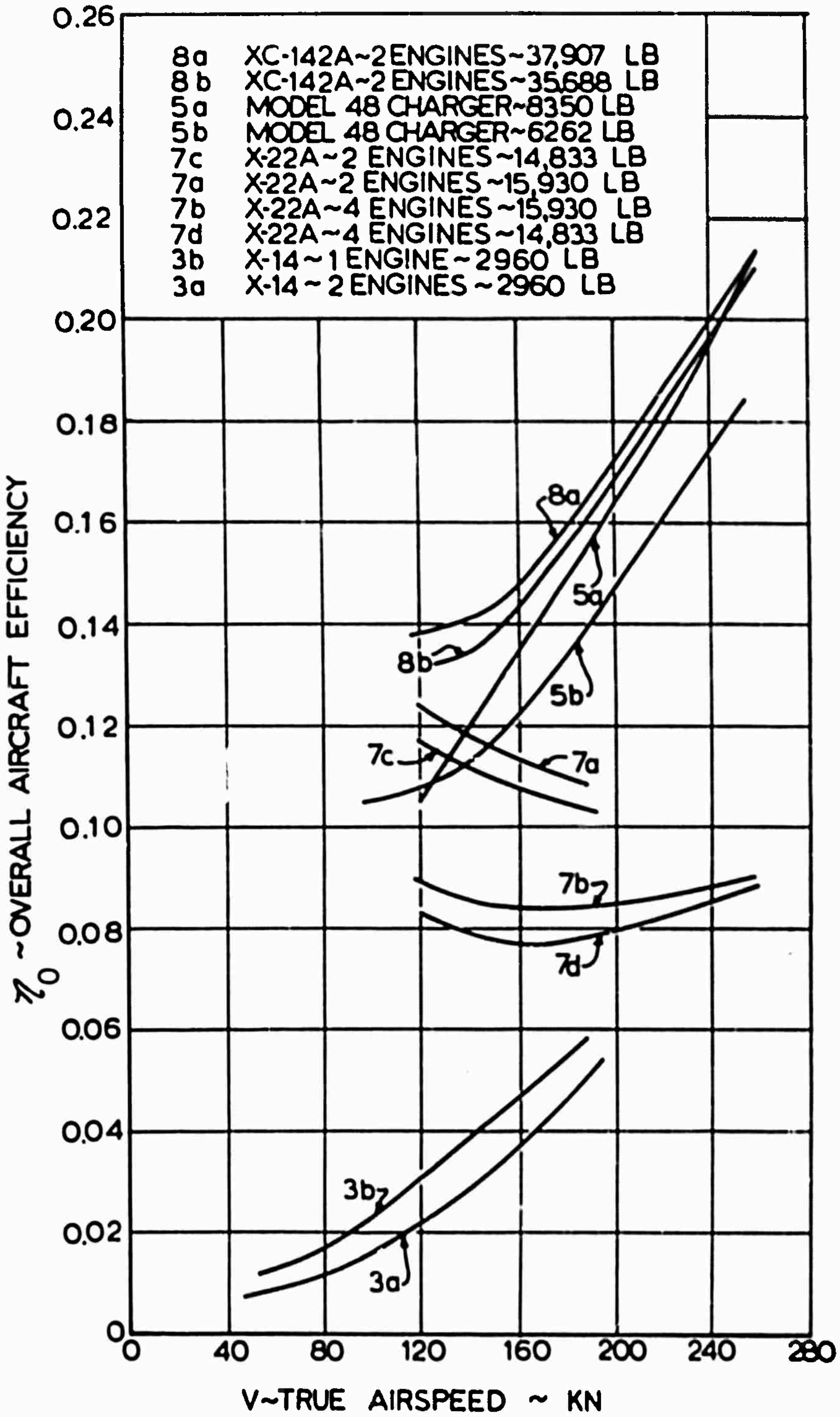

Figure 17. Variation of Overall Efficiency of Four Different Types of Aircraft for Various Engine Operating Conditions and Gross Weights - Sea Level, Standard Day, Level Flight, Normal Rated Power.

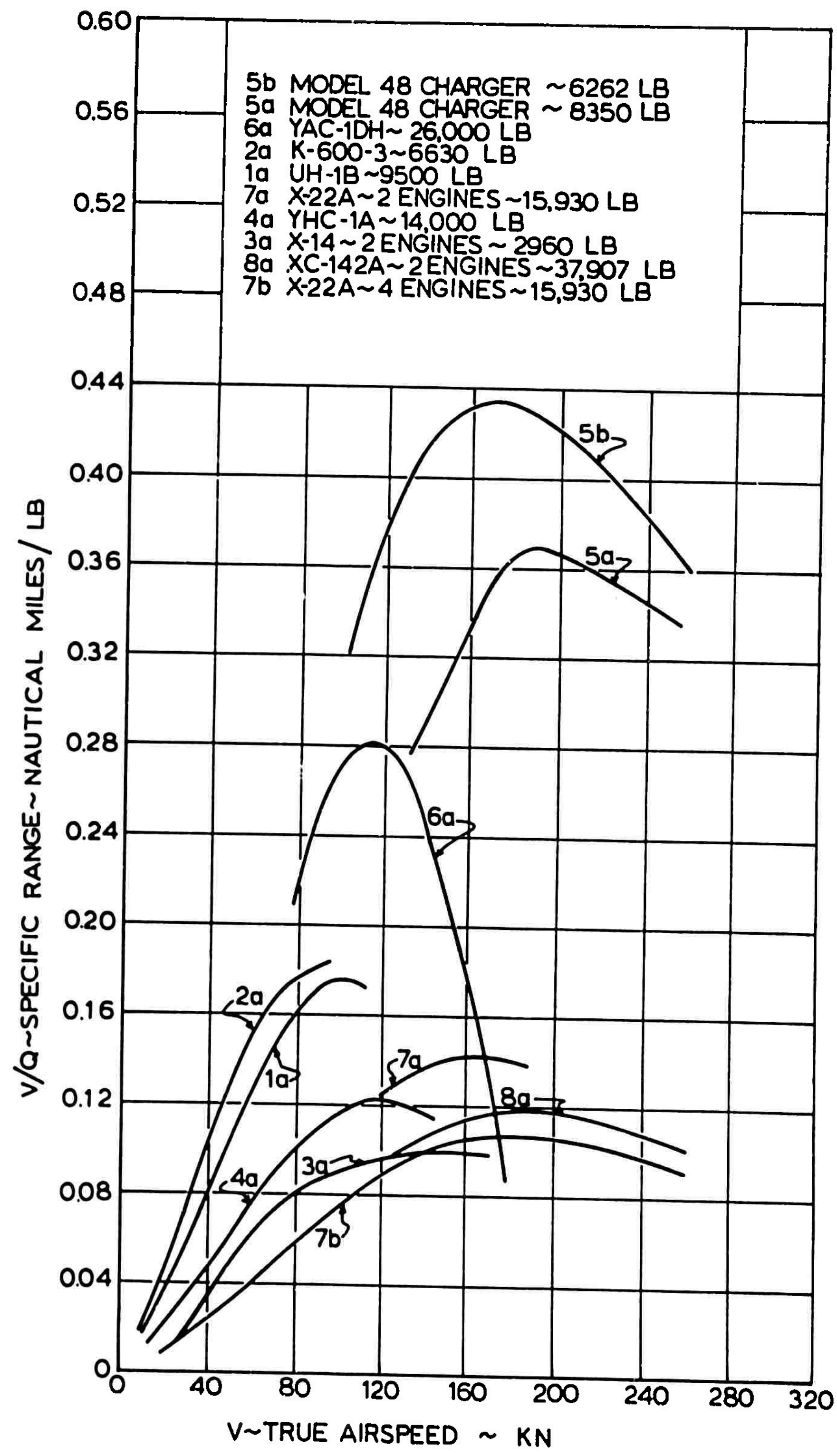

Figure 18. Comparison of the Specific Range of Aircraft as a Function of True Airspeed - Sea Level, Standard Day, Level Flight, Normal Rated Power.

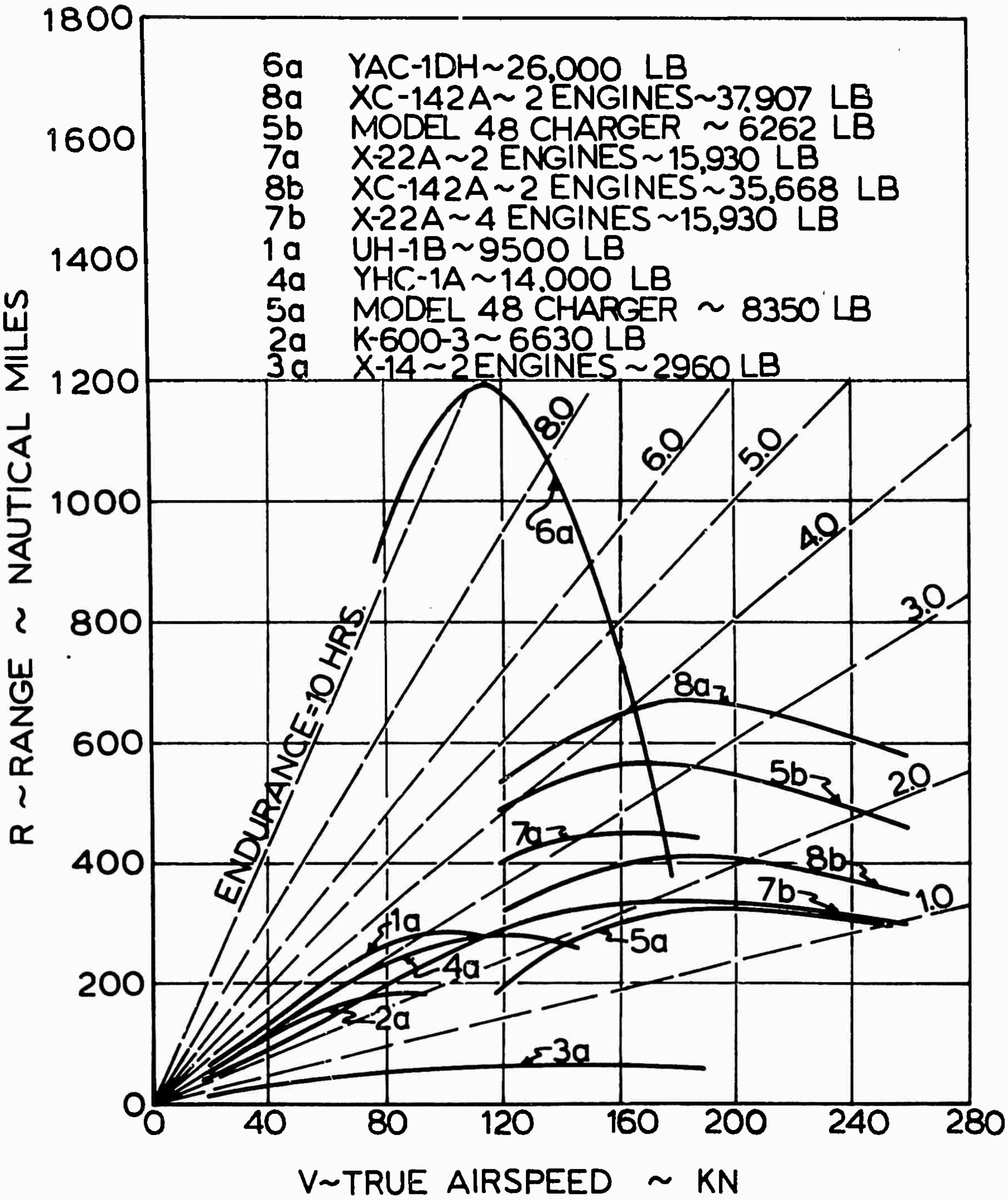

Figure 19. The Effect of True Airspeed on the Range and Endurance Characteristics of Various Aircraft - Sea Level, Standard Day, Level Flight, Normal Rated Power.

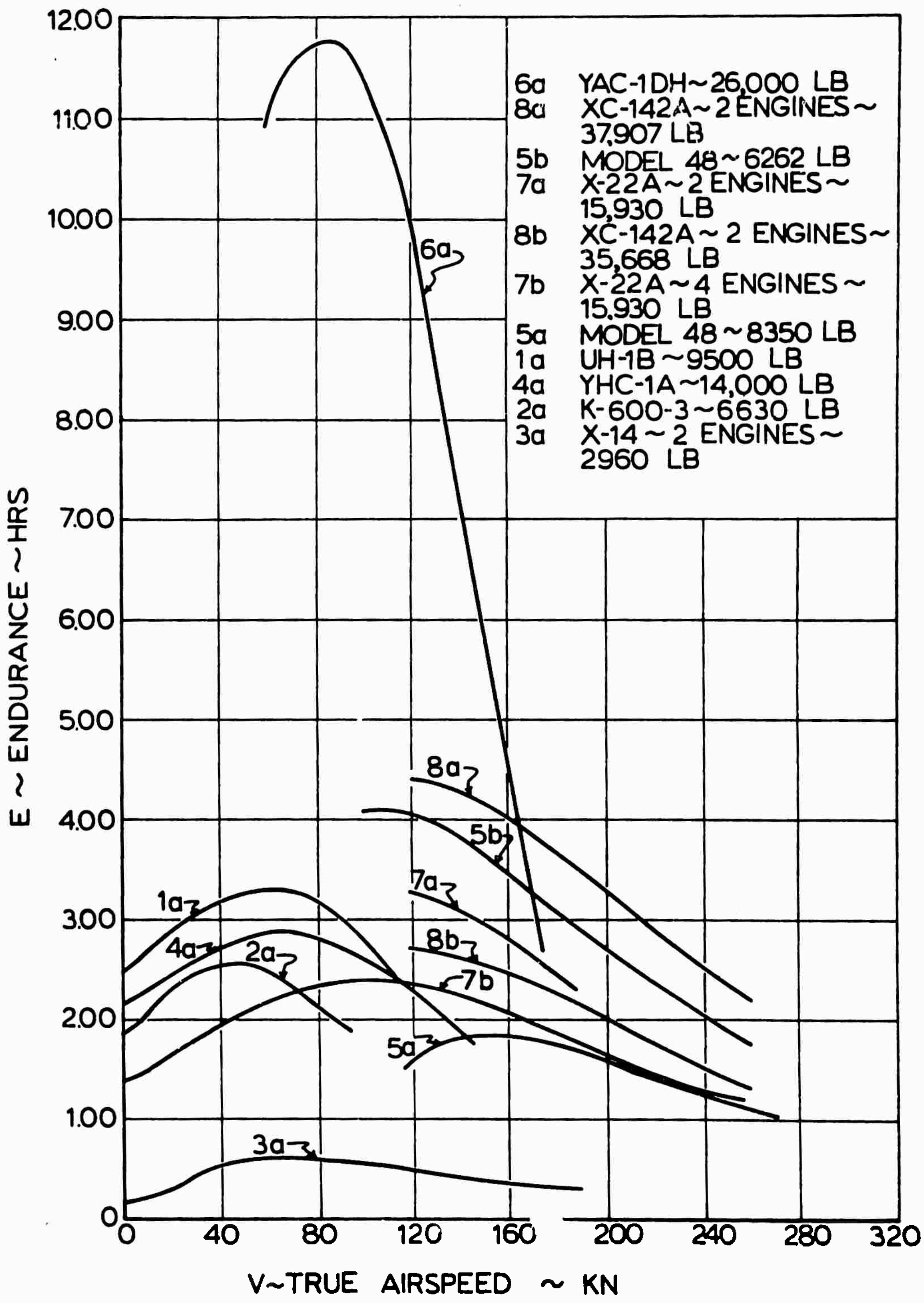

Figure 20. Comparison of the Endurance Capability of Aircraft of Various Configurations in Level Flight - Sea Level, Standard Day, Normal Rated Power.

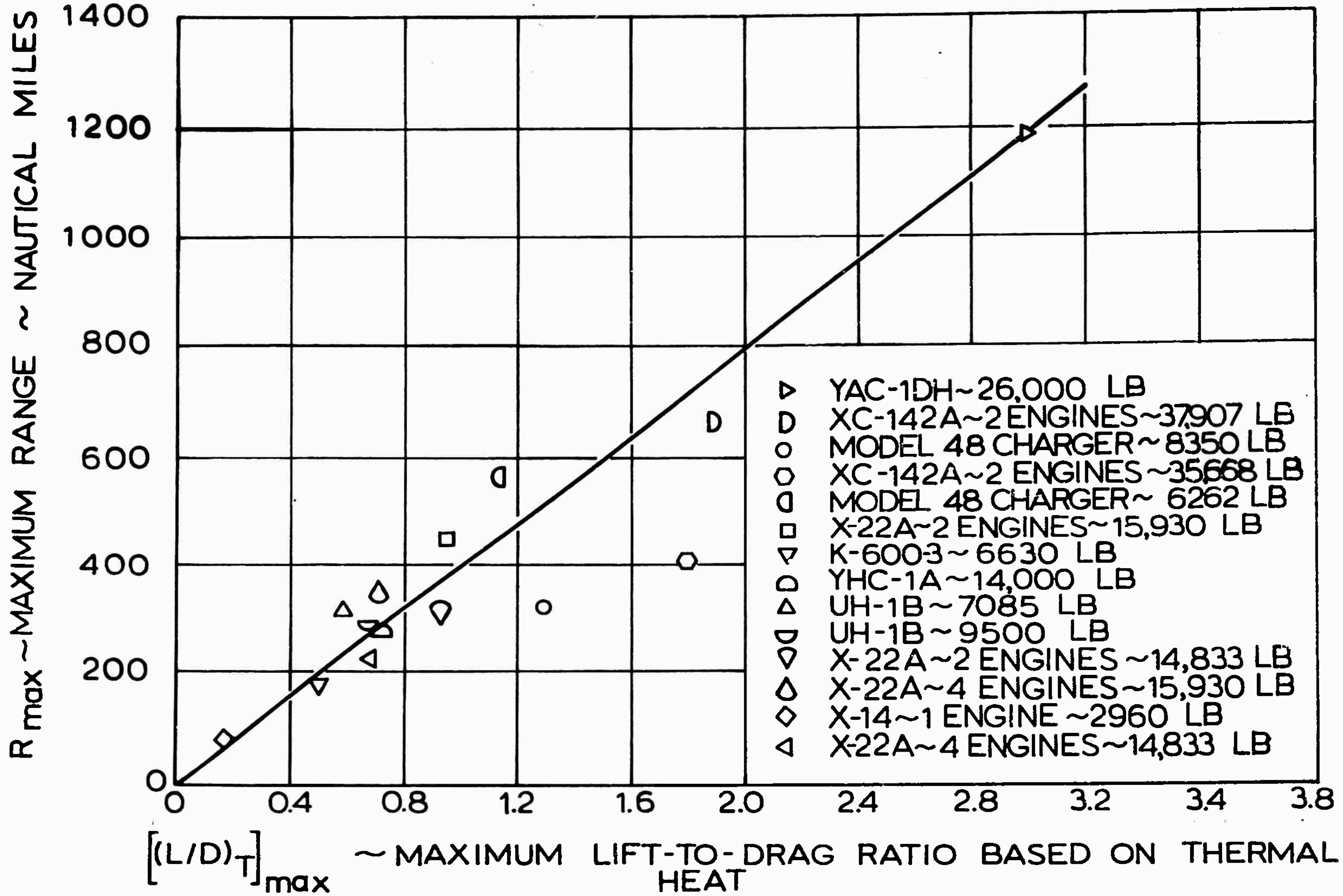

Figure 21. Comparison of Maximum Range and Thermal Lift-to-Drag Ratio of Different Types of Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

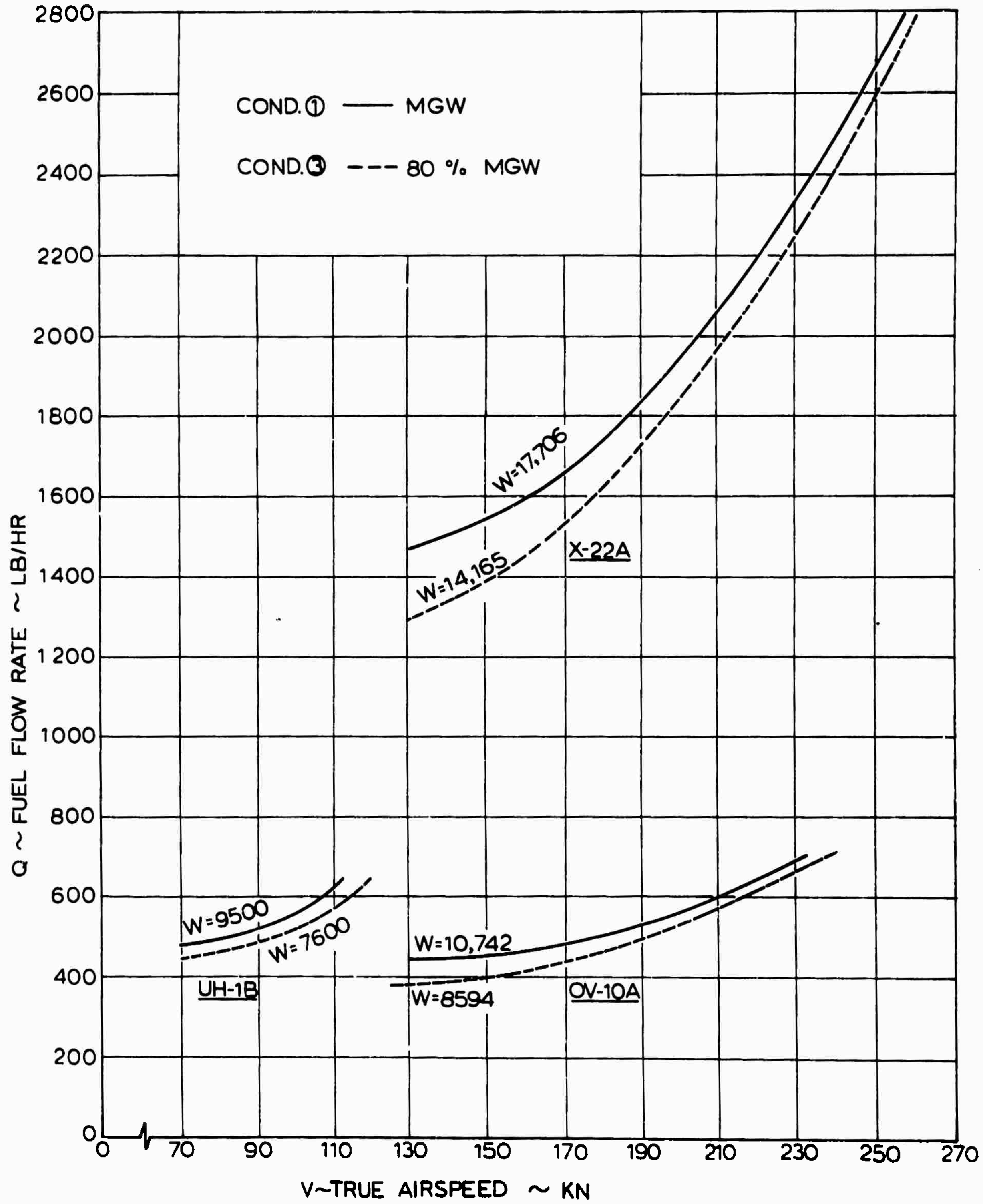

Figure 22. The Effects of Reduced Gross Weight and Airspeed on the Fuel Flow Rate of the UH-1B, X-22A, and OV-10A A_rcraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

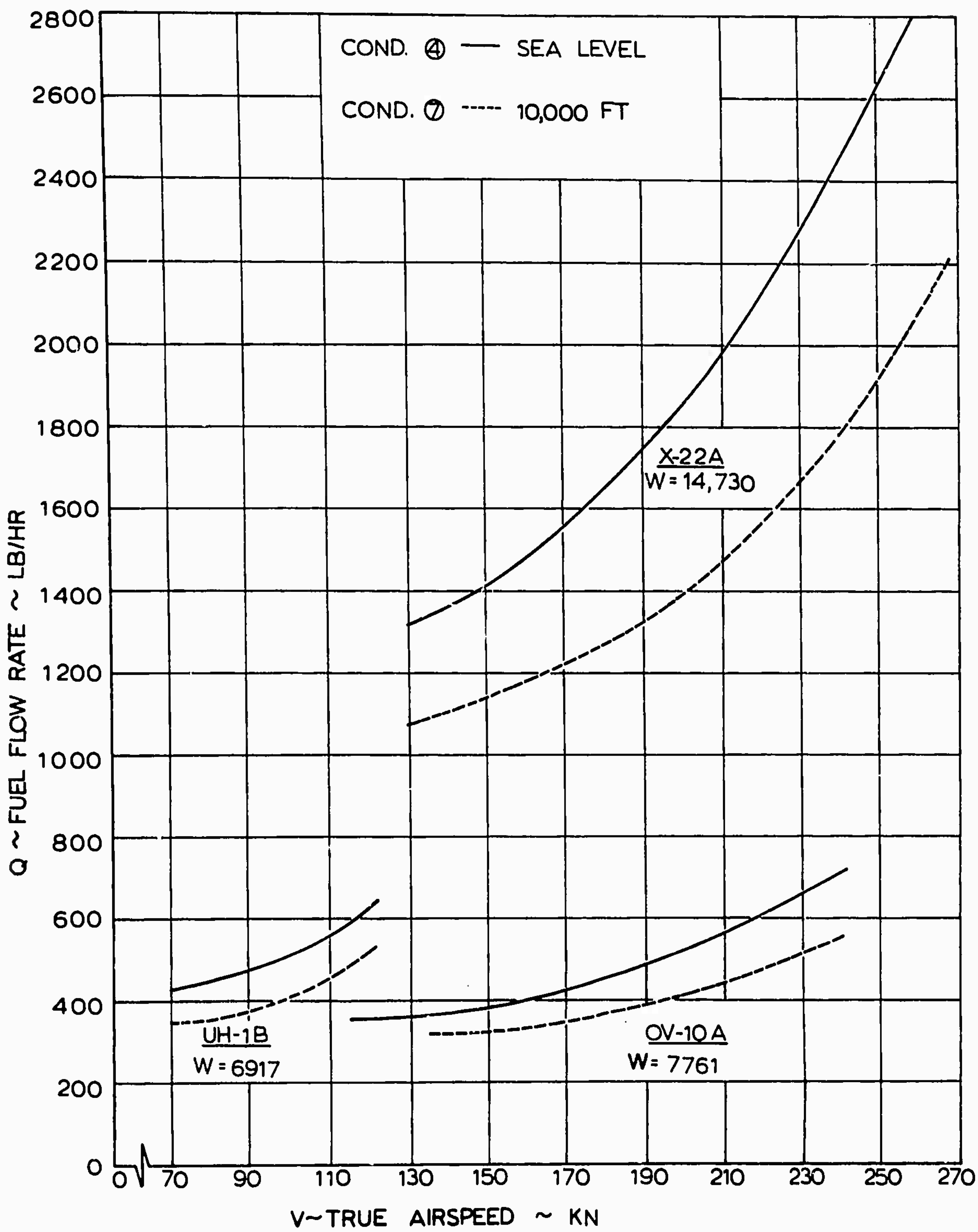

Figure 23. Altitude and Airspeed Effects on the Fuel Flow Rate of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Standard Day, Normal Rated Power.

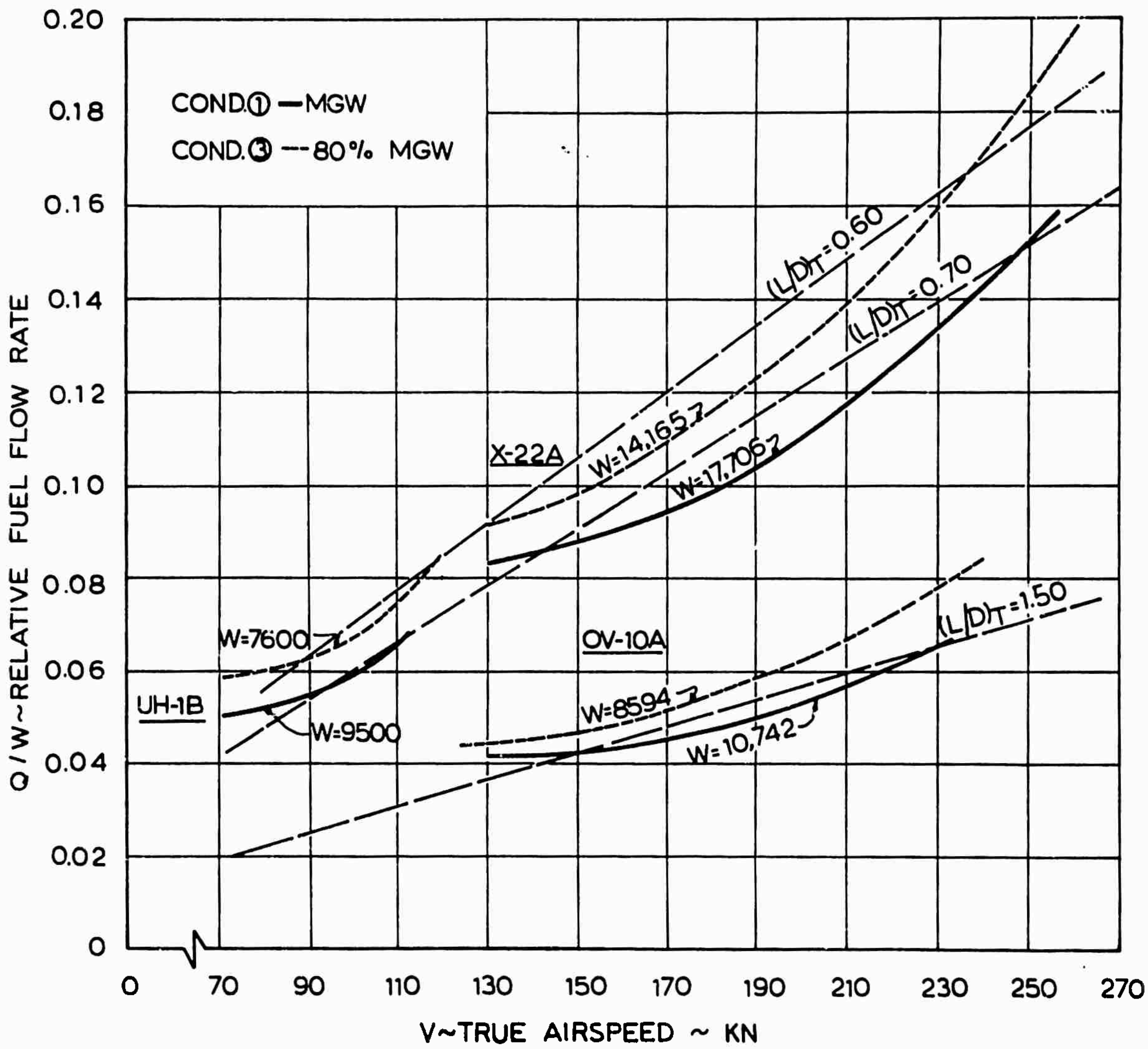

Figure 24. The Effects of Reduced Gross Weight and Airspeed on the Relative Fuel Flow Rate of the UH-1B, X-22A, and OV-10A Aircraft - Sea Level, Level Flight, Standard Day, Normal Rated Power.

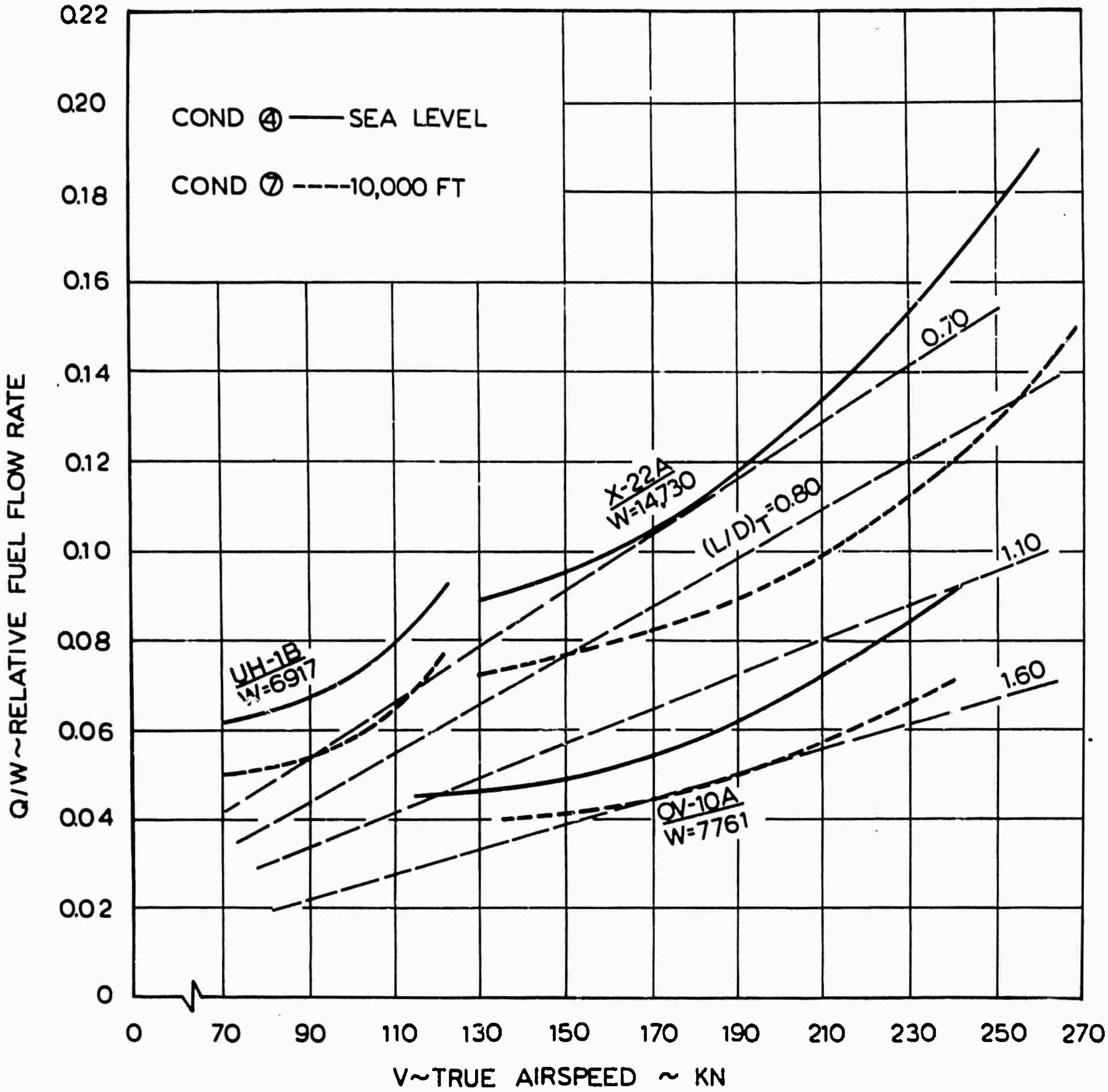

Figure 25. Relative Fuel Flow Rate of the UH-1B, X-22A, and OV-10A Aircraft as a Function of Altitude and Airspeed - Level Flight, Standard Day, Normal Rated Power.

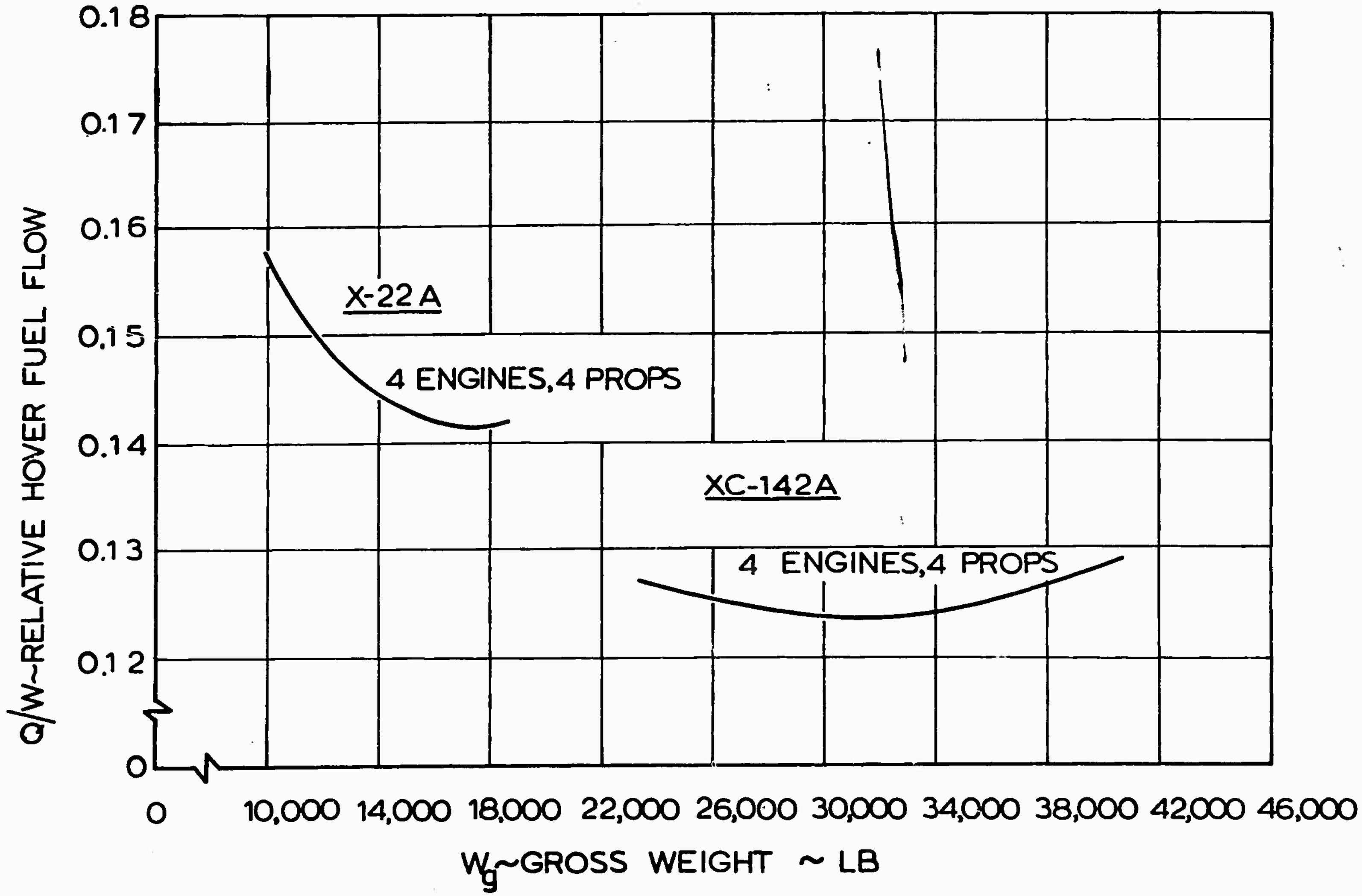

Figure 26. The Effect of Gross Weight on the Relative Hovering Fuel Flow Rate of Two V/STOL Aircraft at Sea Level on a Standard Day.

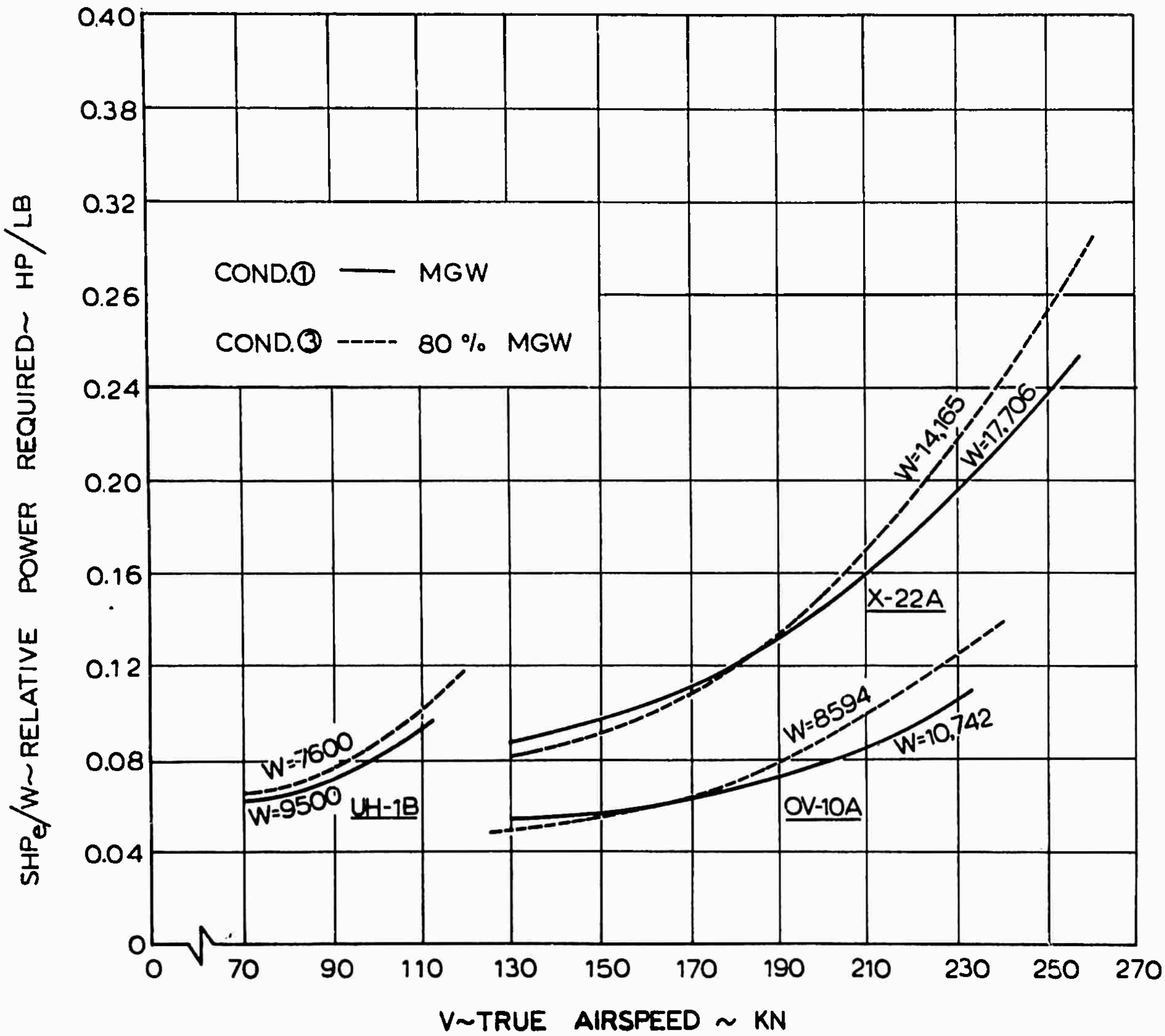

Figure 27. The Effect of Reduced Gross Weight on the Relative Power Requirements of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

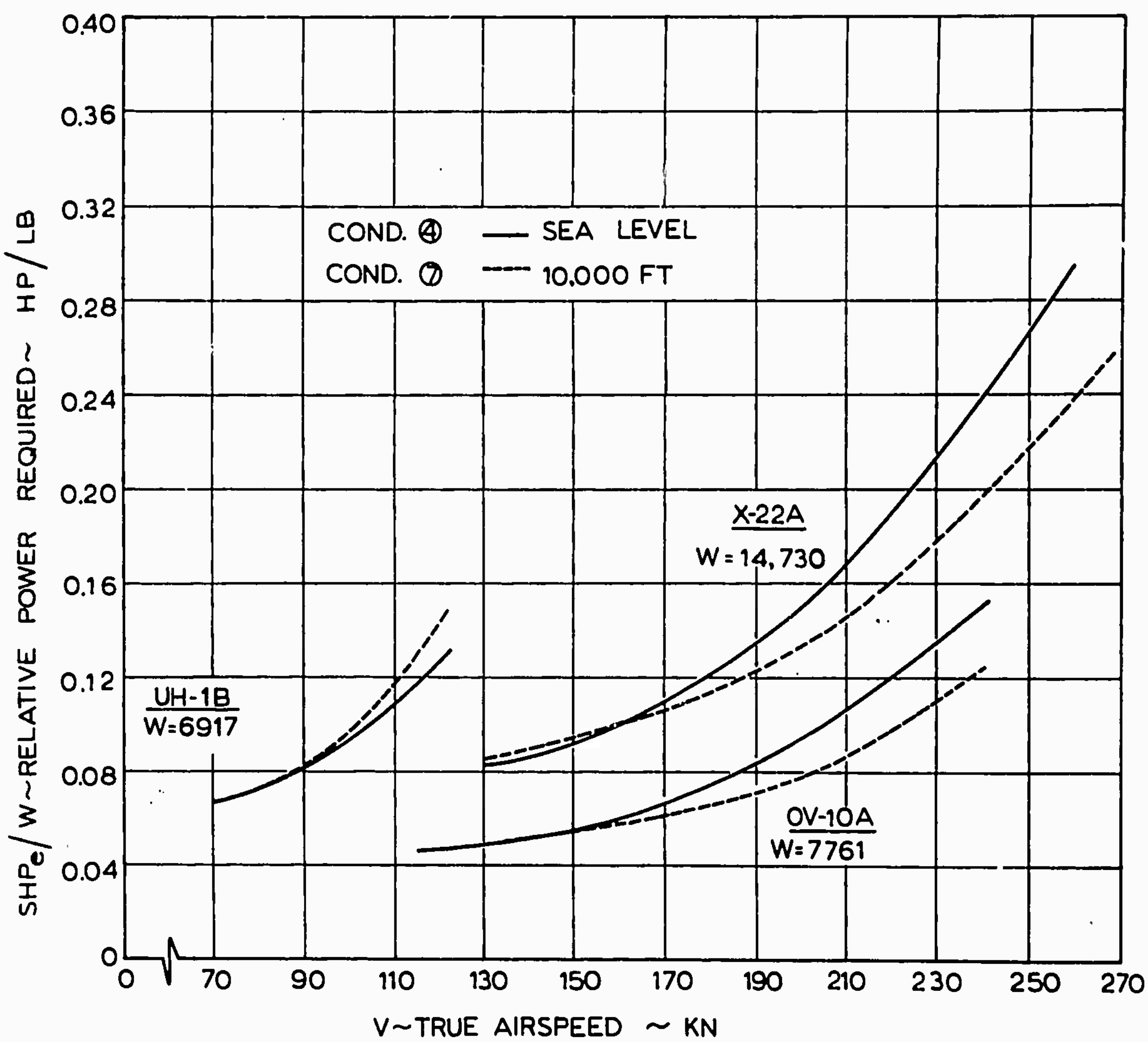

Figure 28. Comparison of the Relative Power Required by the UH-1B, OV-10A, and X-22A at Sea Level and at an Altitude of 10,000 Feet - Level Flight, Standard Day, Normal Rated Power.

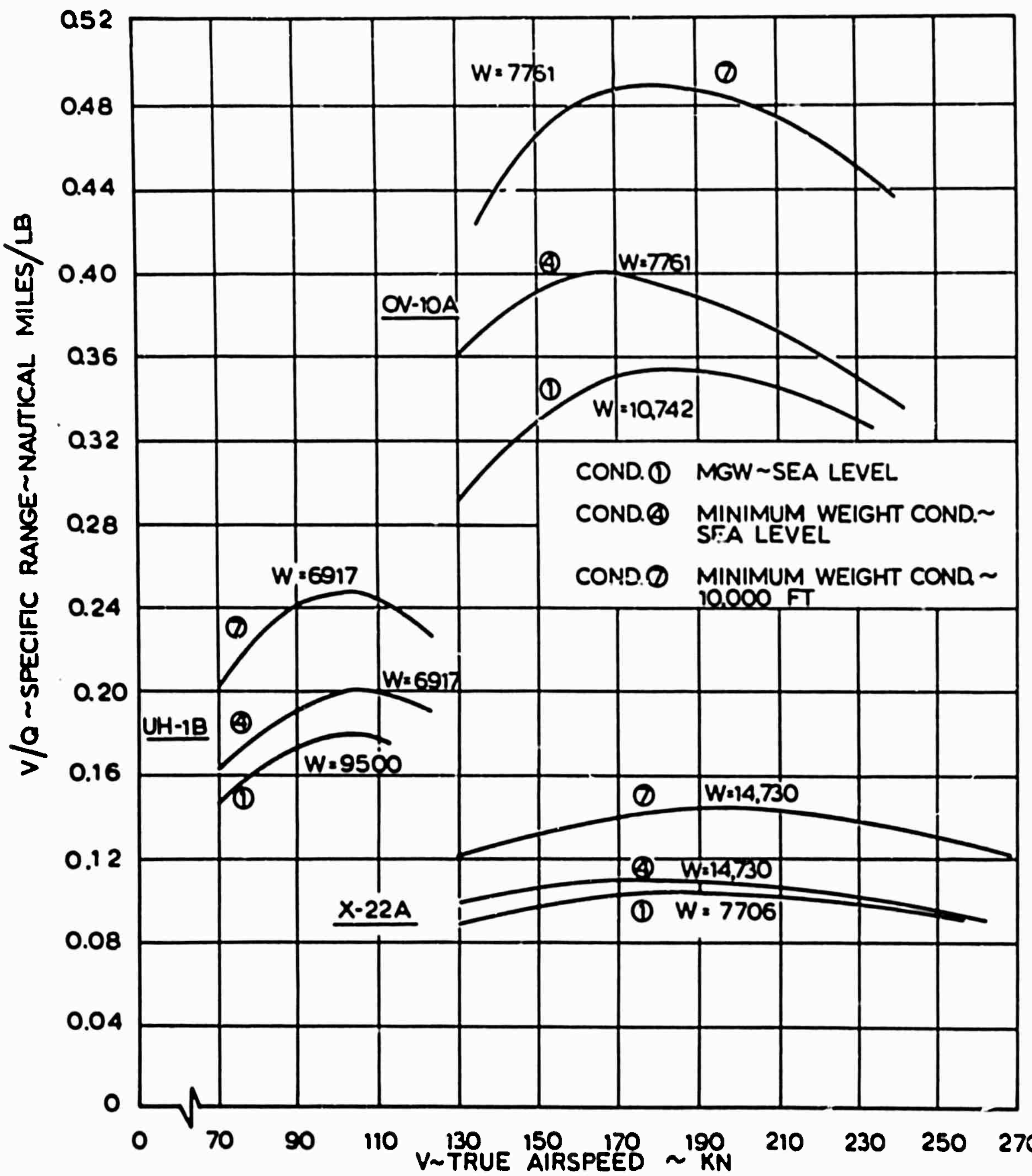

Figure 29. Variation of Specific Range of the UH-1B, OV-10A, and X-22A Aircraft With Changes of Altitude and Gross Weight - Level Flight, Standard Day, Normal Rated Power.

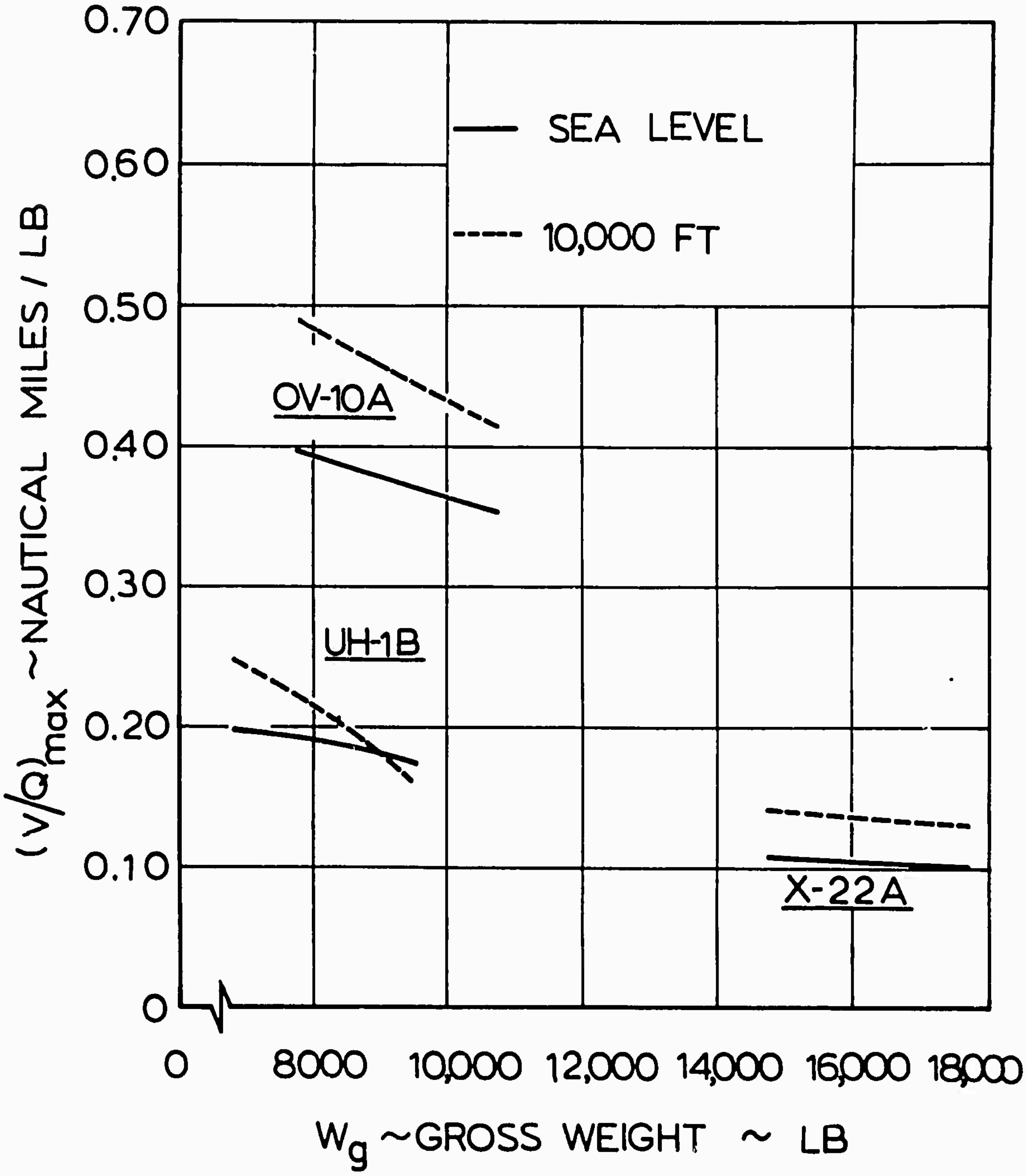

Figure 30. The Effects of Altitude and Gross Weight on the Maximum Specific Range of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Standard Day, Normal Rated Power.

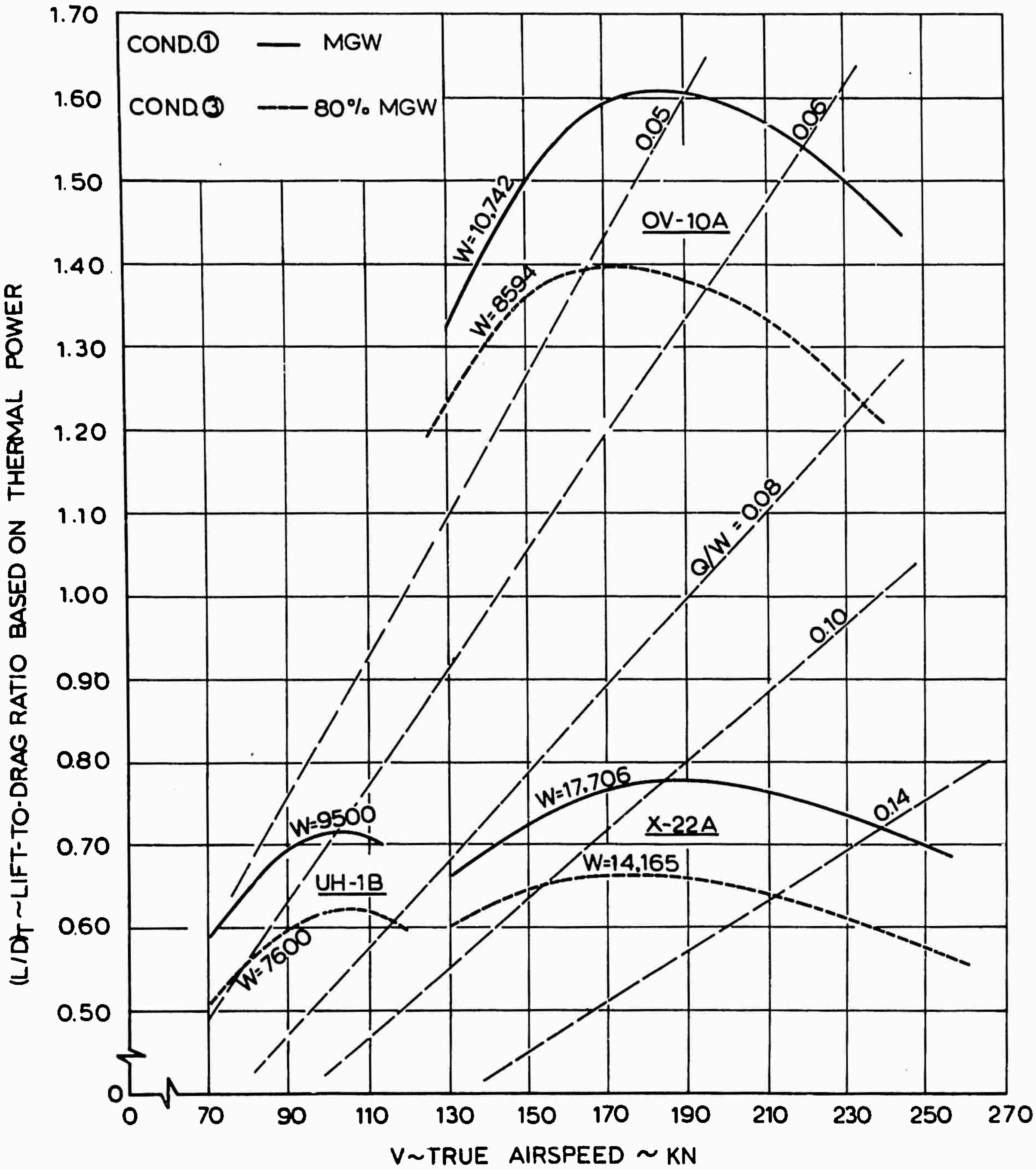

Figure 31. Gross Weight Effect on the Thermal Power Characteristics of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

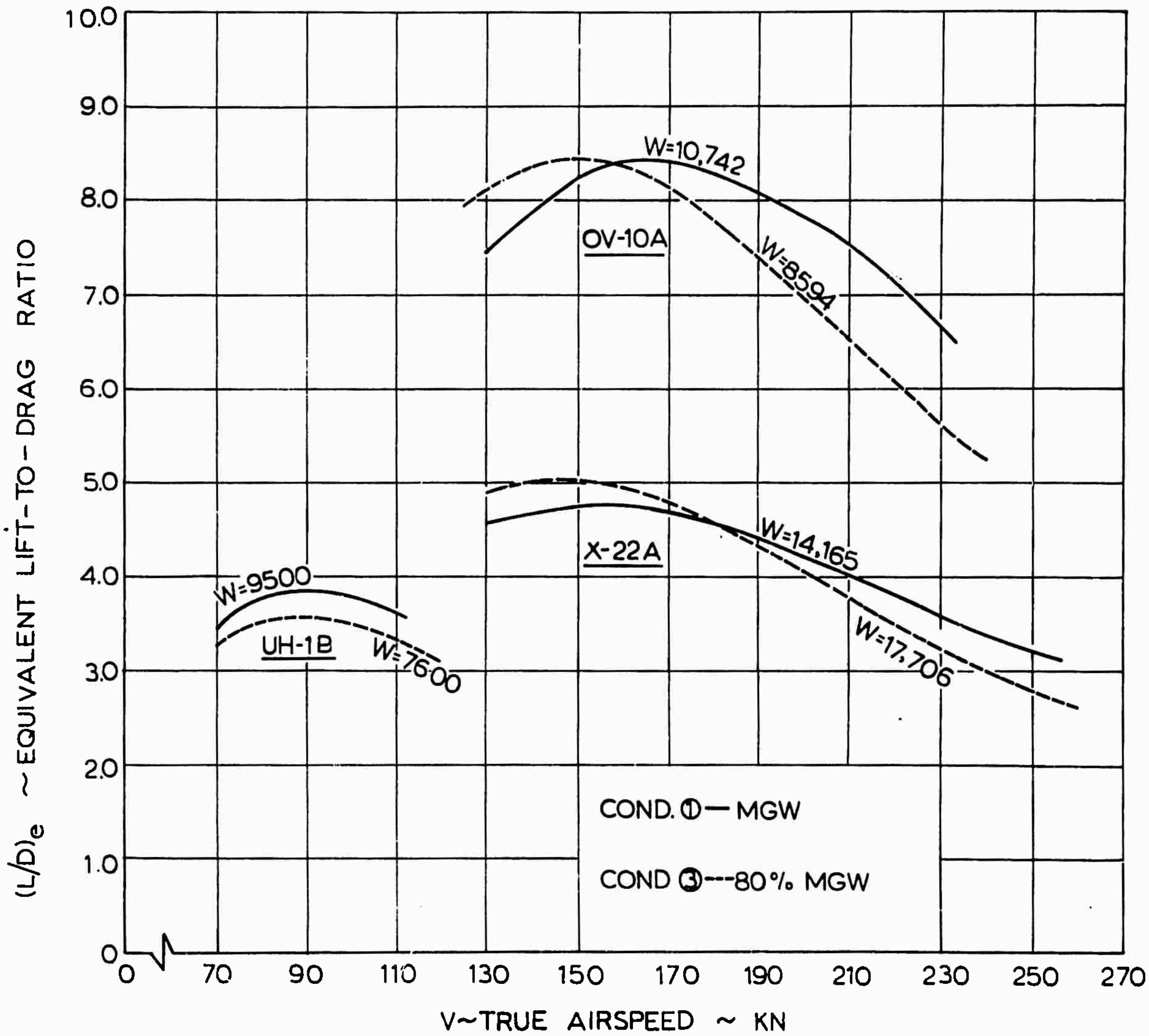

Figure 32. Gross Weight Effect on the Equivalent Lift-to-Drag Ratio of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

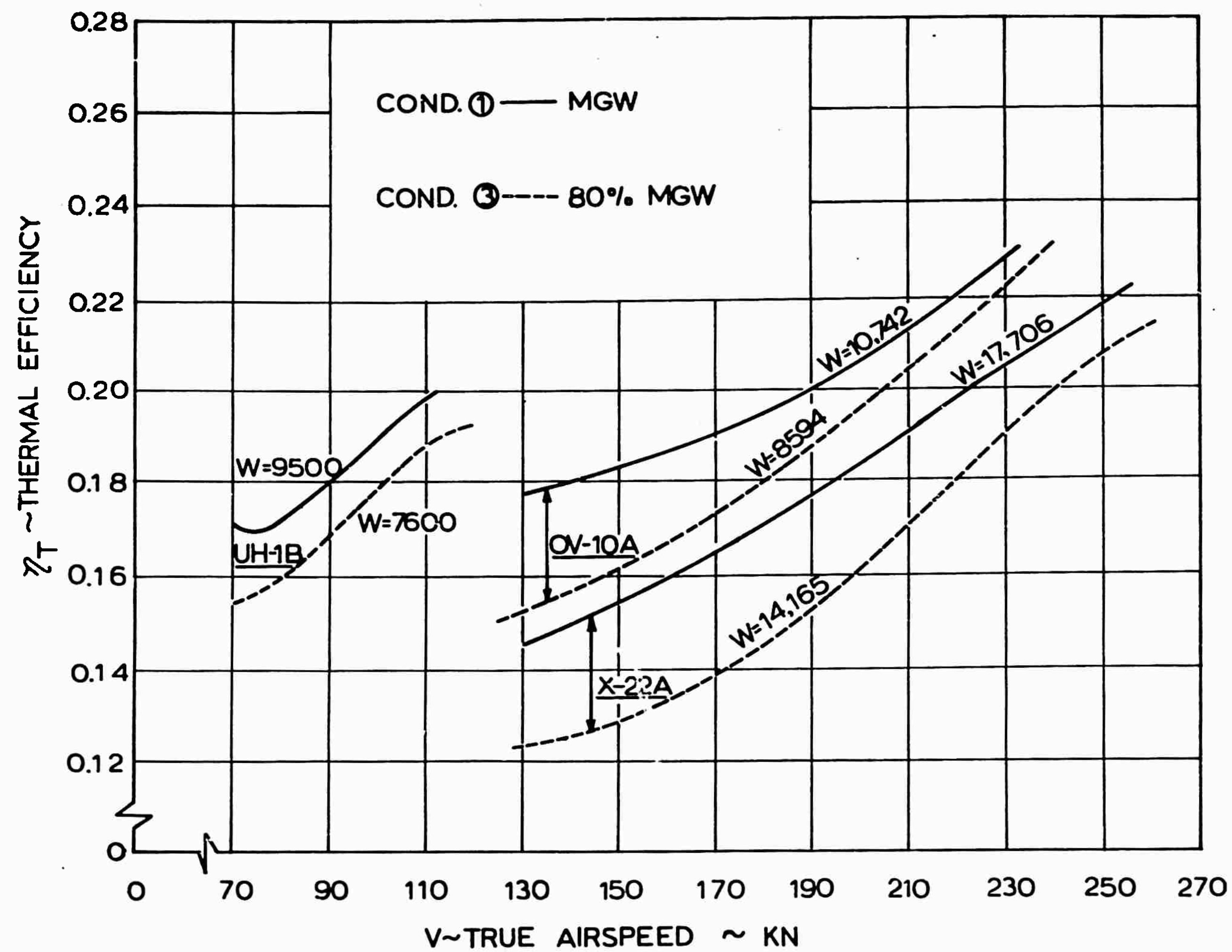

Figure 33. The Effect of Gross Weight on the Thermal Efficiency of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Sea Level, Standard Day, Normal Rated Power.

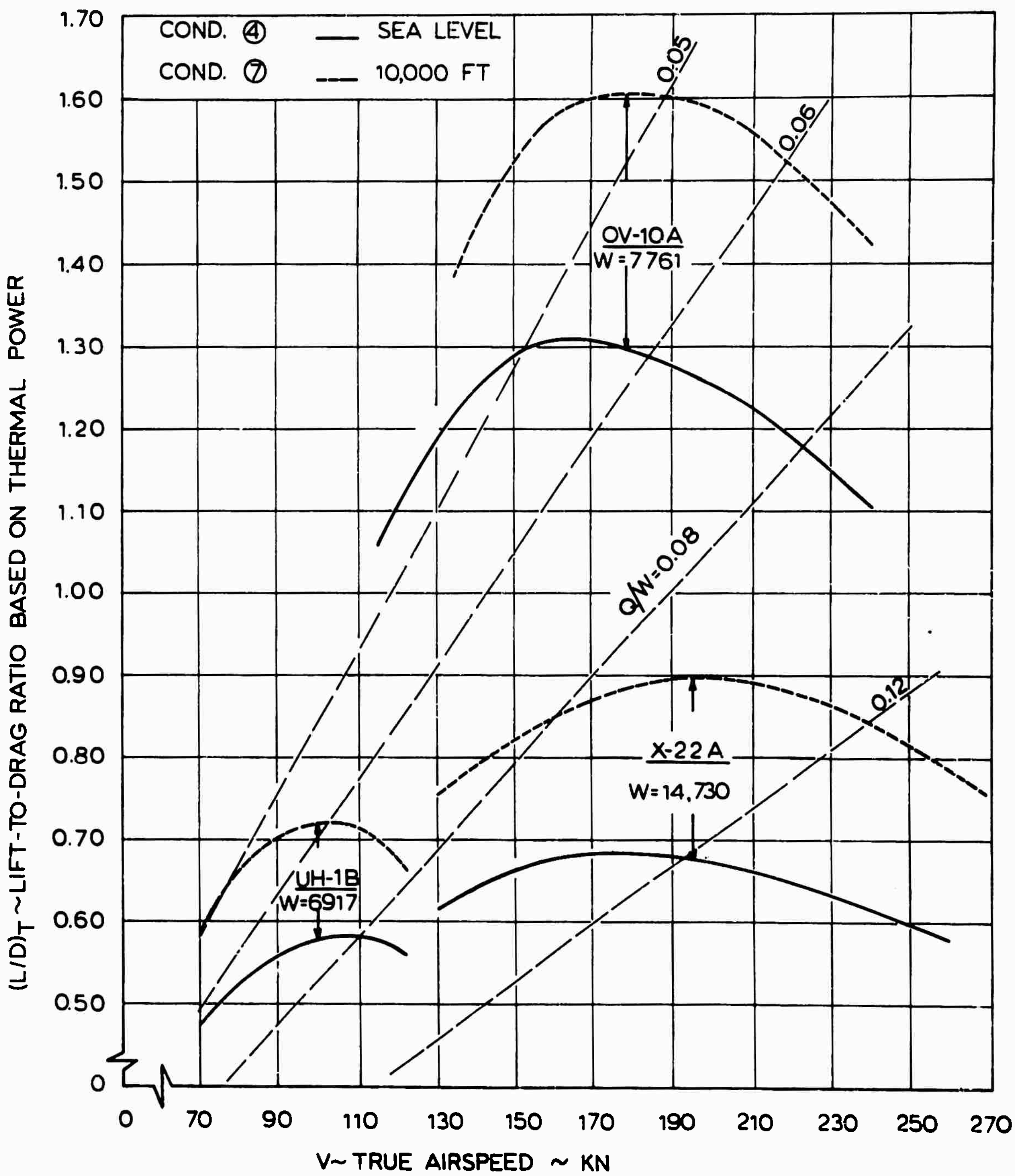

Figure 34. Thermal Lift-to-Drag Ratio of the UH-1B, OV-10A, and X-22A Aircraft as a Function of Altitude - Level Flight, Standard Day, Normal Rated Power.

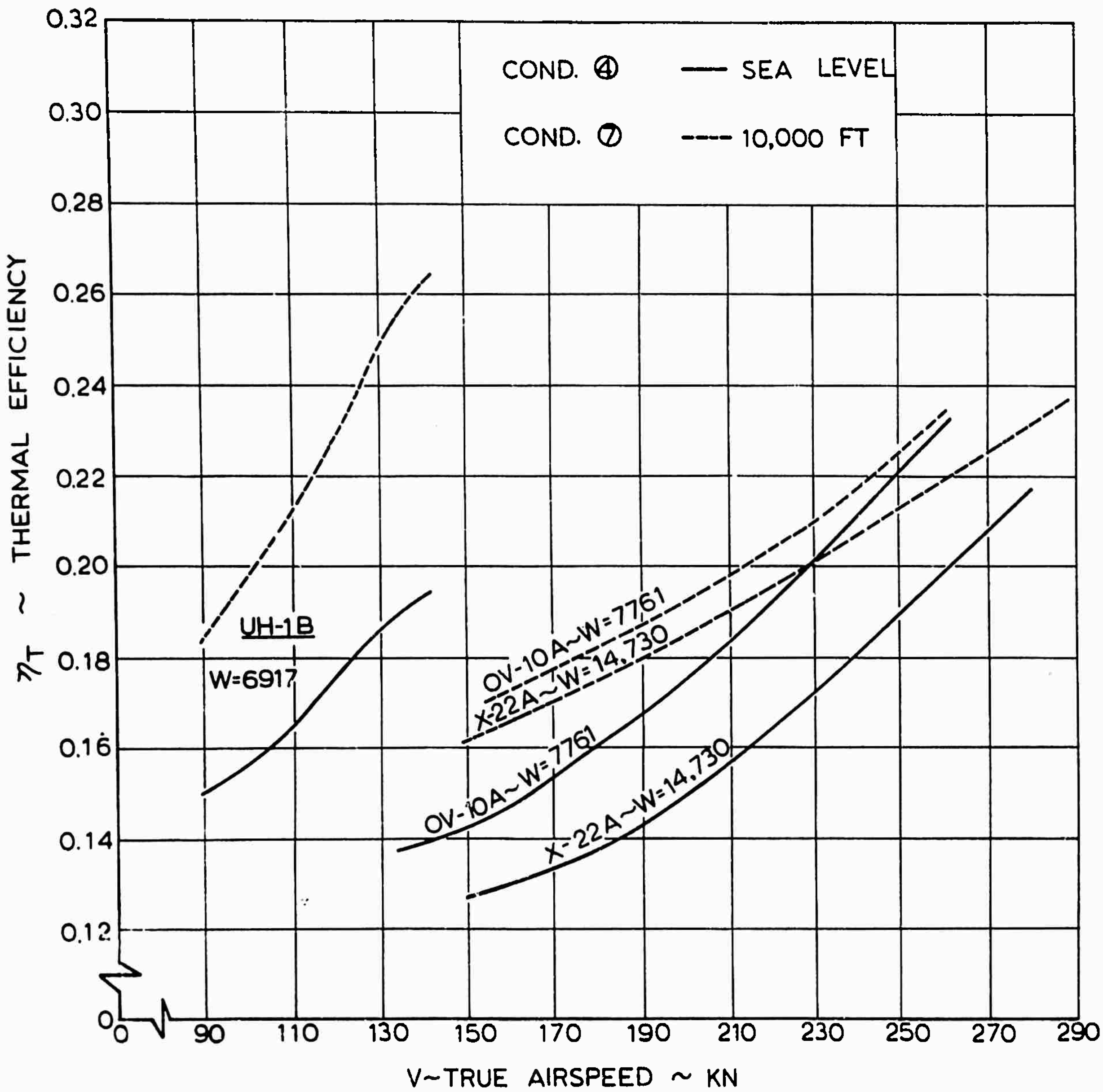

Figure 35. The Effects of Altitude and Airspeed on the Thermal Efficiency of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Standard Day, Normal Rated Power.

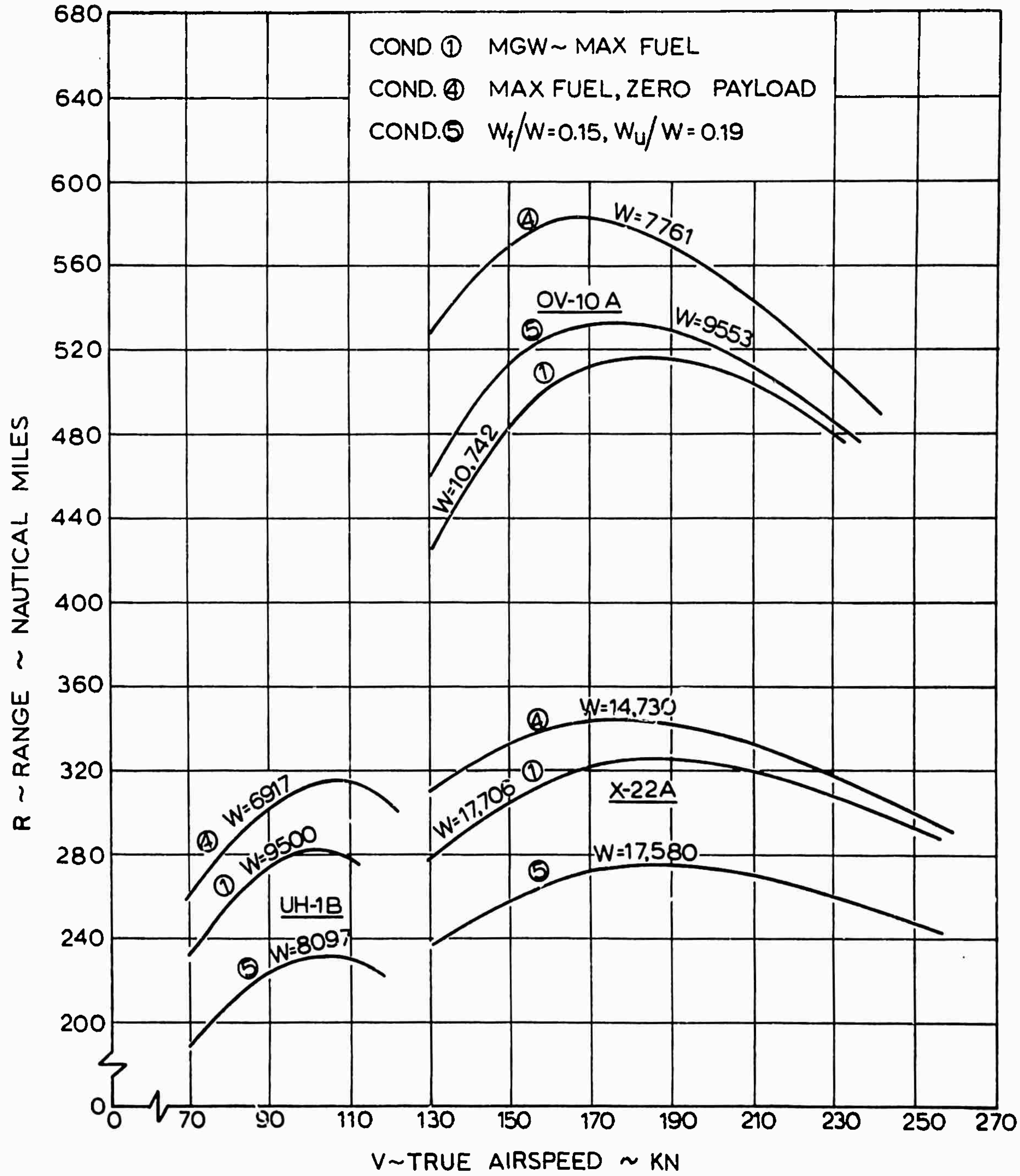

Figure 36. Range Characteristics of the UH-1B, OV-10A, and X-22A Aircraft With Various Fuel Loads and Payloads - Level Flight, Sea Level, Standard Day, Normal Rated Power.

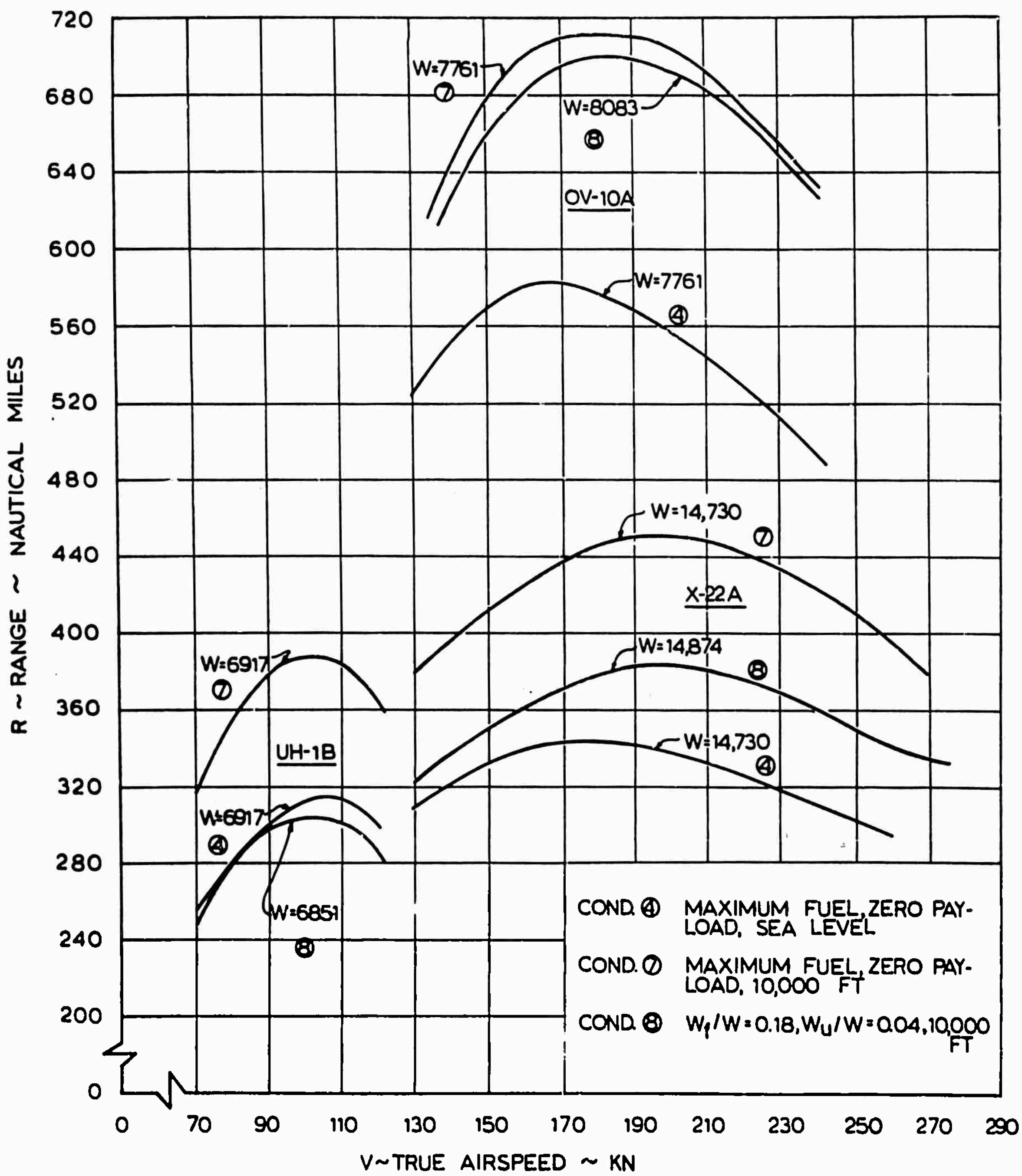

Figure 37. Range Characteristics of the UH-1B, OV-10A, and X-22A Aircraft for Various Altitude and Loading Conditions - Full Internal Fuel, Level Flight, Standard Day, Normal Rated Power.

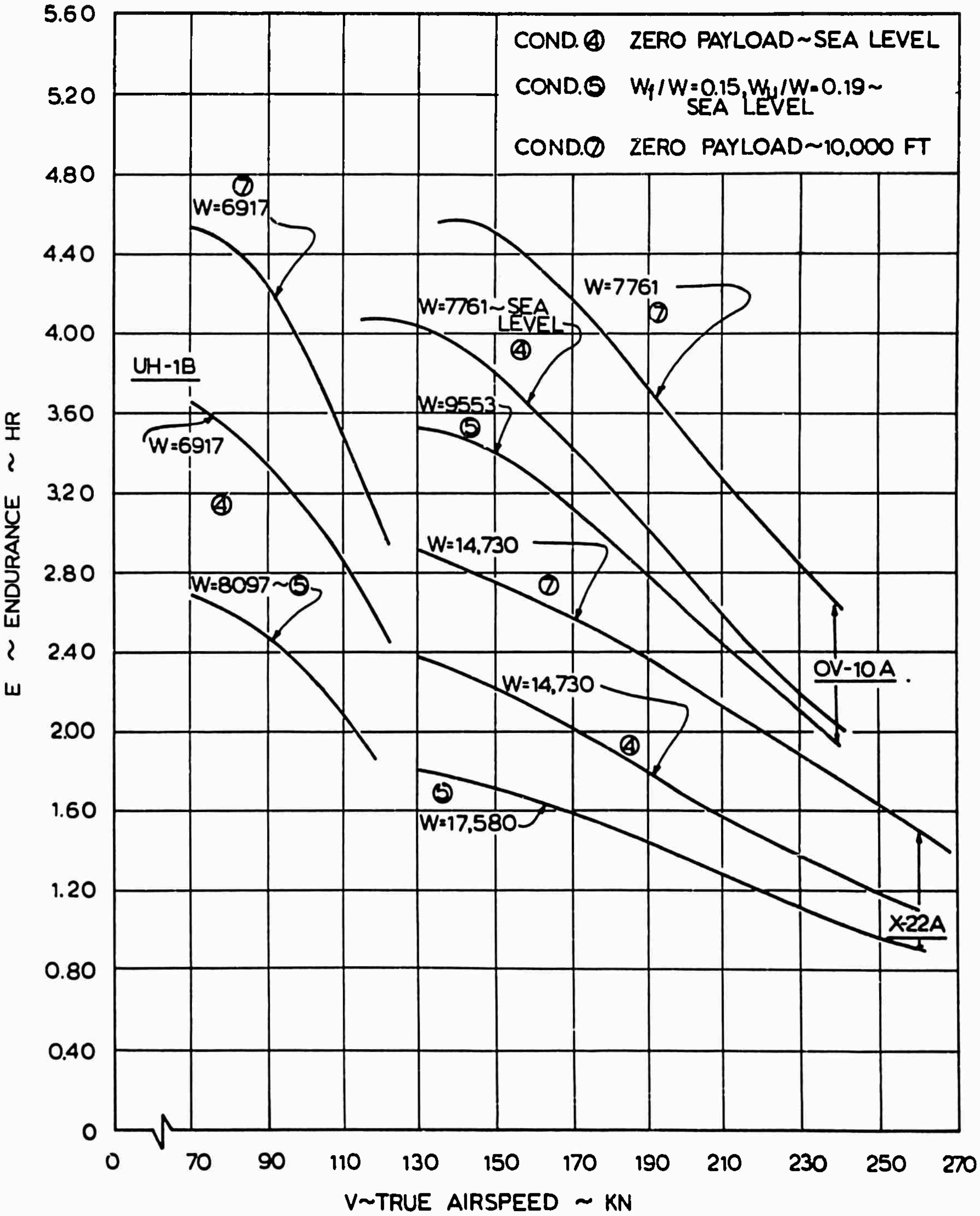

Figure 38. A Comparison of the Endurance Characteristics of the UH-1B, OV-10A, and X-22A Aircraft as a Function of Altitude and for Equal Relative Fuel Load and Payload Conditions - Level Flight, Standard Day, Normal Rated Power.

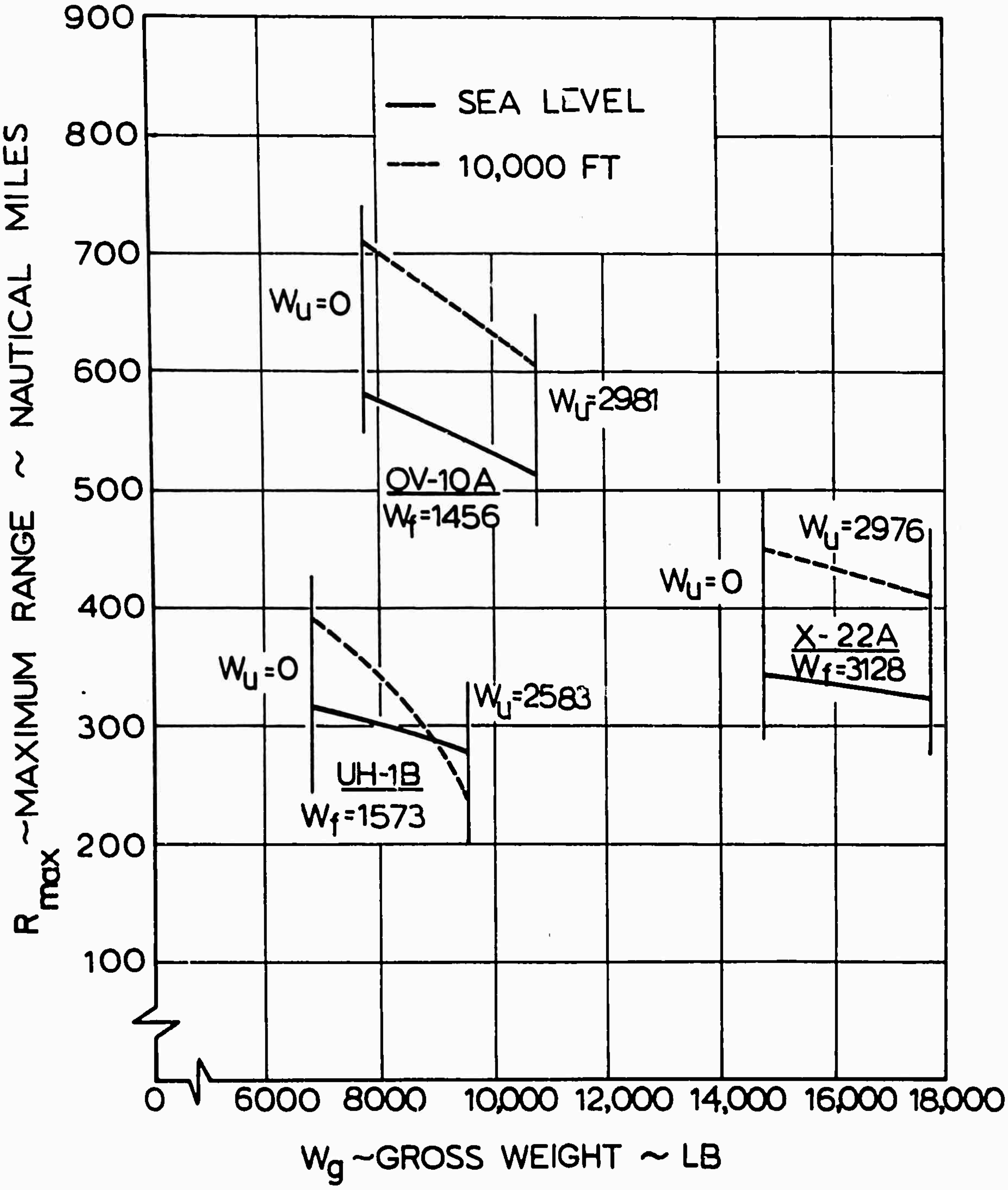

Figure 39. Maximum Range Comparison of the UH-1B, OV-10A, and X-22A Aircraft With Full Internal Fuel and Varying Payload - Level Flight, Standard Day, Normal Rated Power.

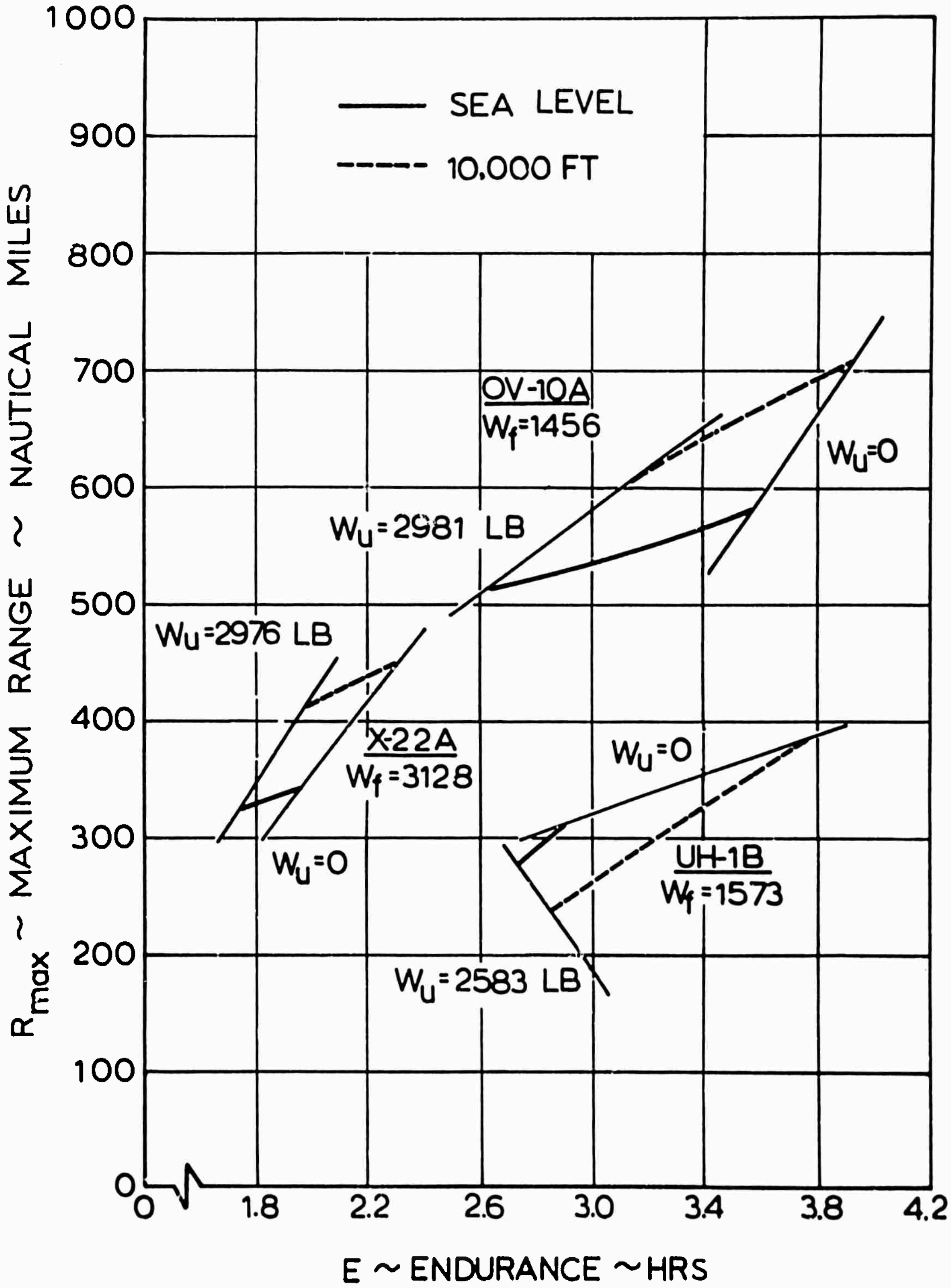

Figure 40. Comparison of Endurance at Maximum Range of the UH-1B, OV-10A, and X-22A Aircraft as a Function of Payload and Altitude - Full Internal Fuel, Level Flight, Standard Day, Normal Rated Power.

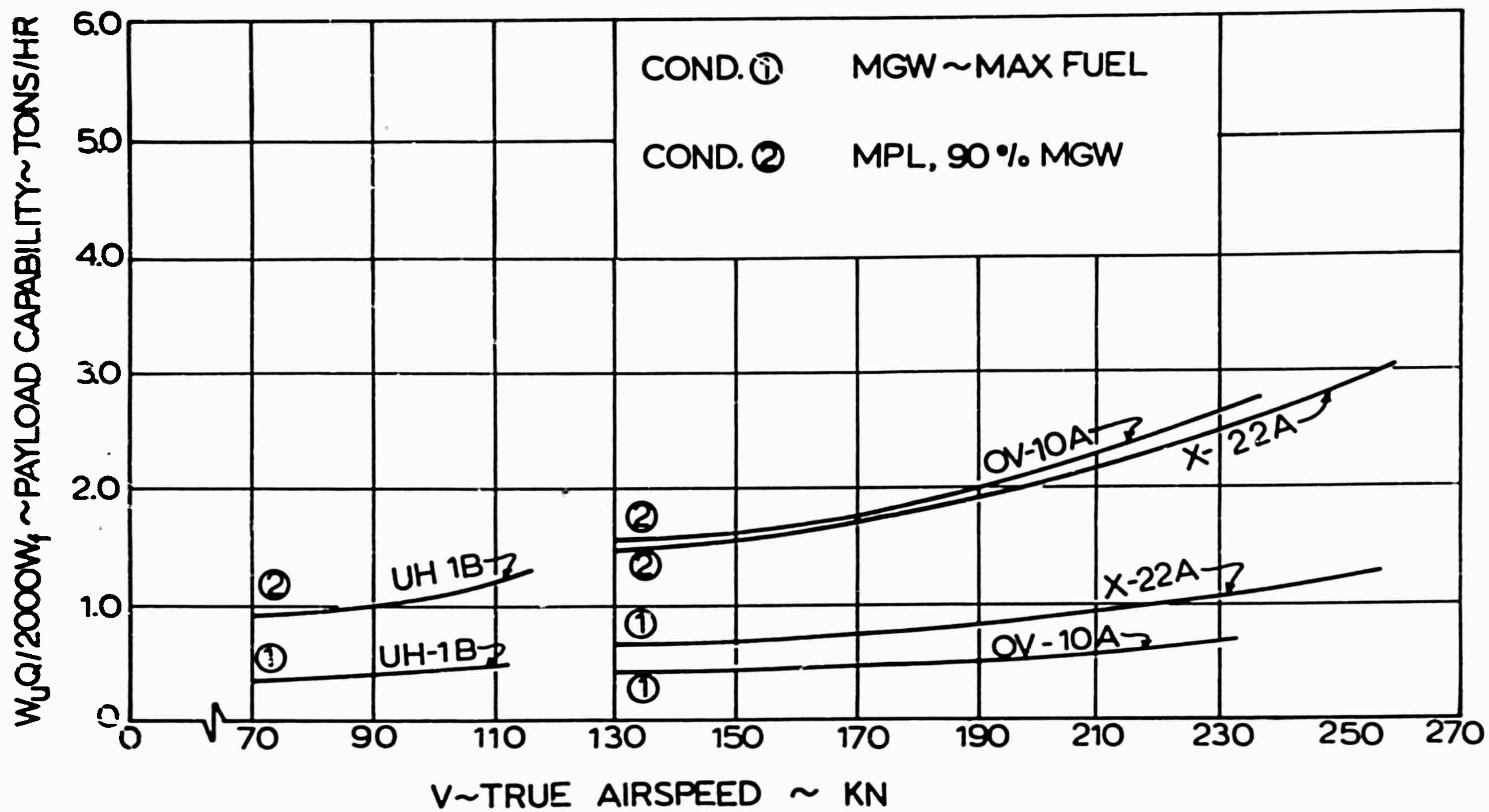

Figure 41. The Effect of Reducing Fuel Load on the Payload Capability of Aircraft While Maintaining Maximum Payload Conditions - Level Flight, Sea Level, Standard Day, Normal Rated Power.

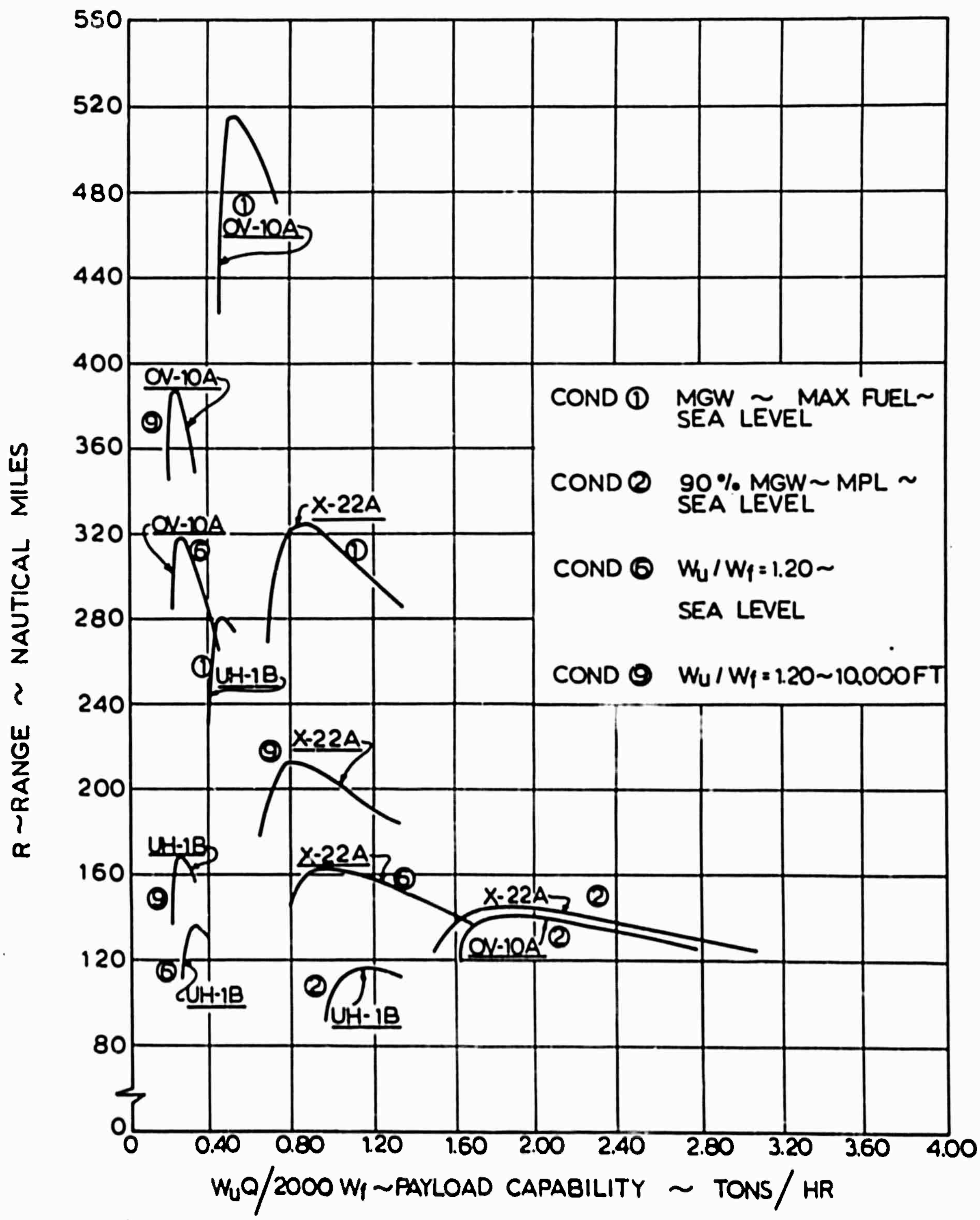

Figure 42. Comparisons of Range vs. Payload Capability Curves of the UH-1B, OV-10A, and X-22A Aircraft for Various Payload and Fuel Load Conditions at Sea Level and 10,000 Feet - Level Flight, Standard Day, Normal Rated Power.

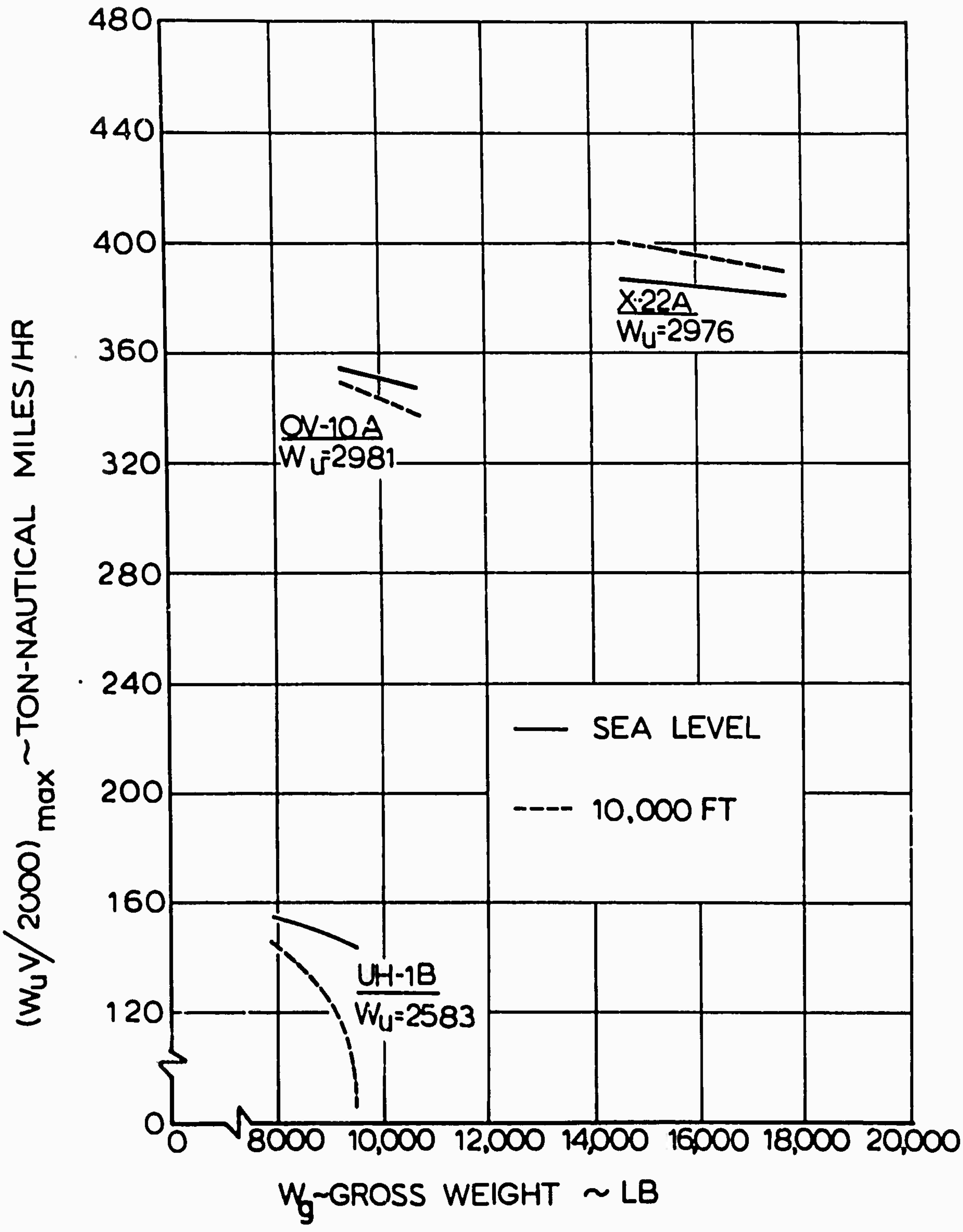

Figure 43. The Effects of Altitude and Gross Weight on the Rate of Payload - Range Capability of the UH-1B, OV-10A, and X-22A Aircraft With Maximum Payload - Level Flight, Standard Day, Normal Rated Power.

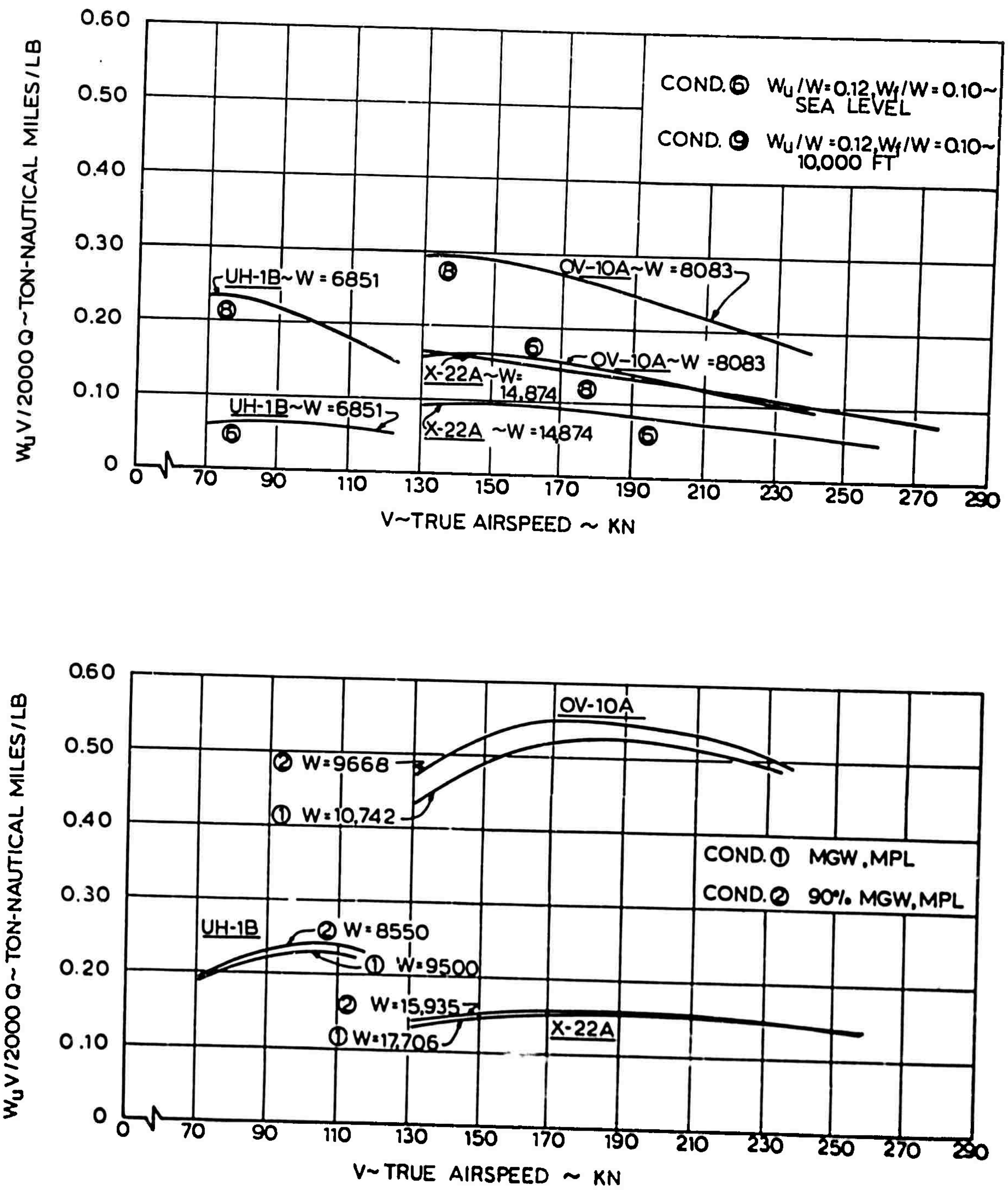

Figure 44. The Effects of Various Loading Conditions and Altitude on the Payload - Range Efficiency of the UH-1B, OV-10A, and X-22A Aircraft - Level Flight, Standard Day, Normal Rated Power.

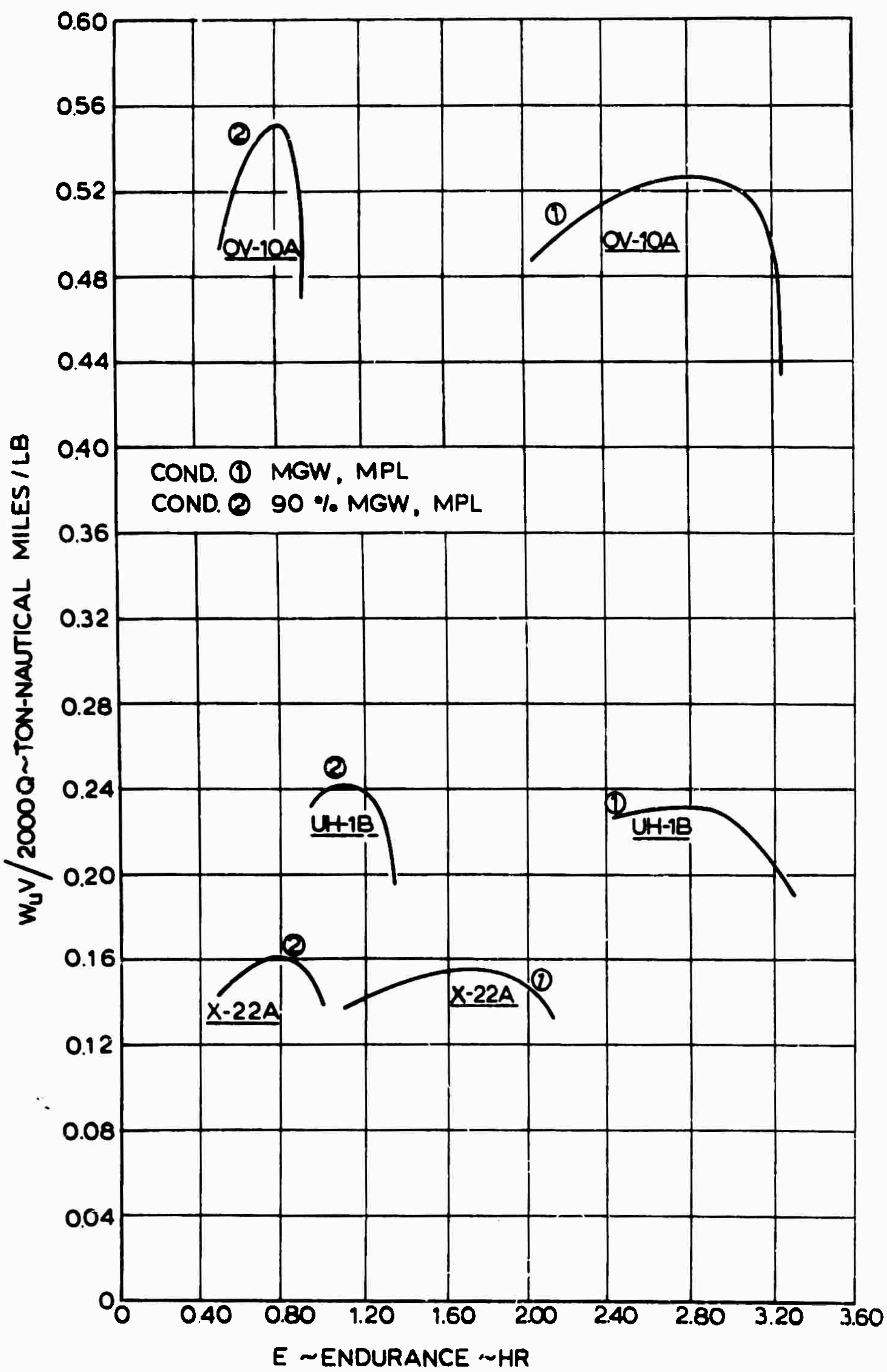

Figure 45. Endurance and Payload - Range Efficiency of the UH-1B, OV-10A, and X-22A Aircraft as Affected by a Reduction of Gross Weight While Maintaining Maximum Payload Conditions - Sea Level, Level Flight, Standard Day, Normal Rated Power.

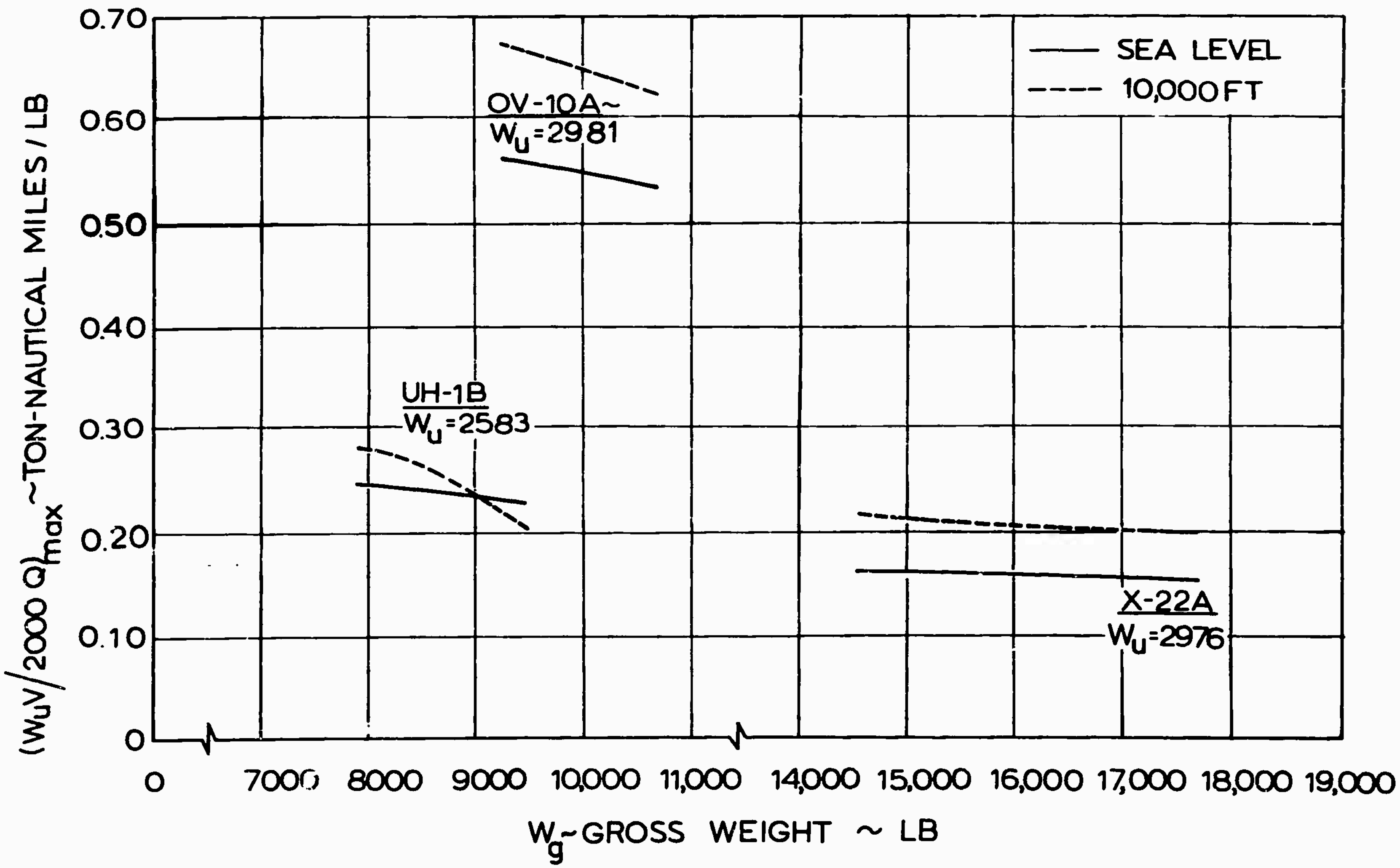

Figure 46. Comparison of Maximum Payload - Range Efficiency of the UH-1B, OV-10A, and X-22A Aircraft With Maximum Payload as a Function of Altitude and Gross Weight - Level Flight, Standard Day, Normal Rated Power.

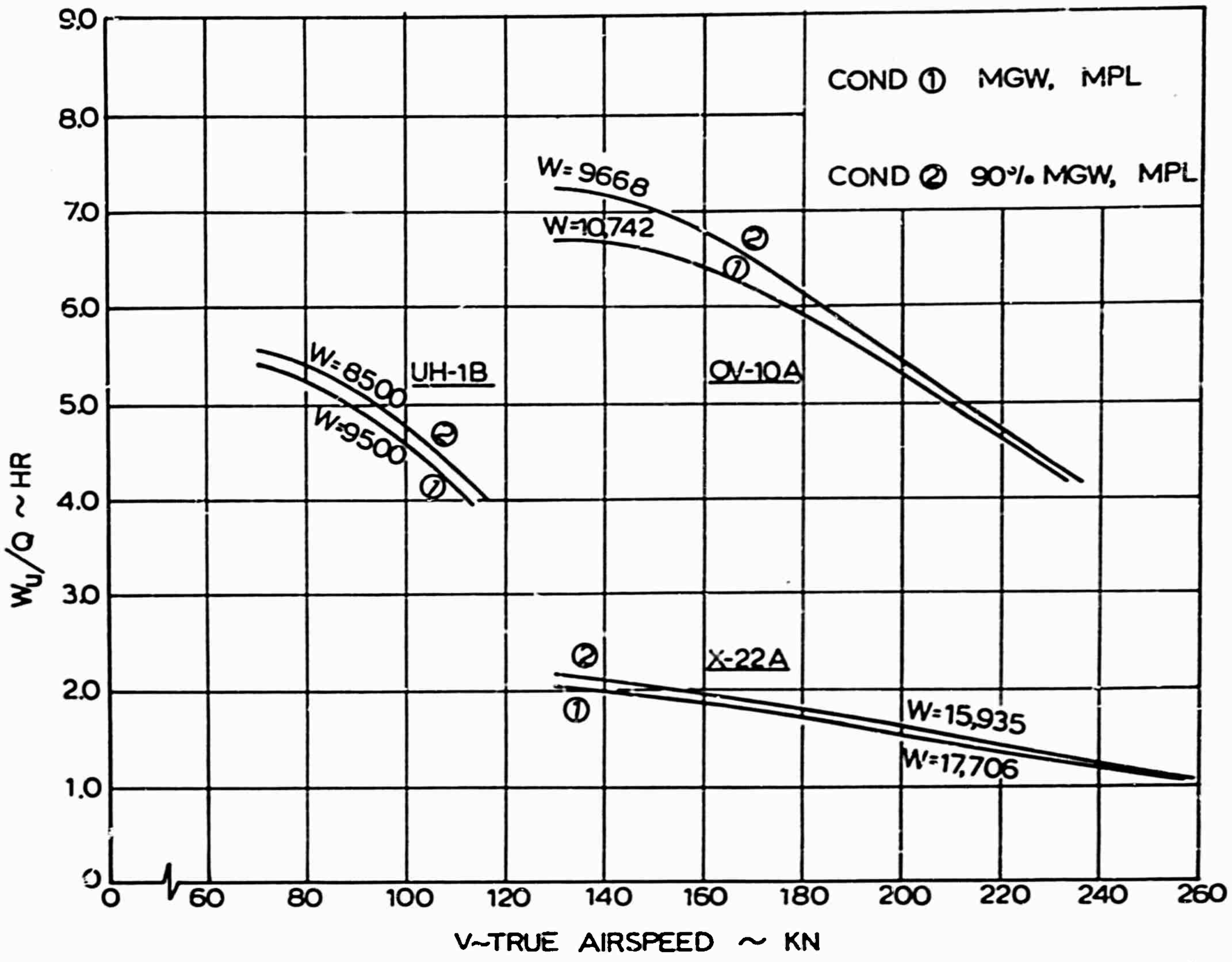

Figure 47. Effect of a Reduction of Gross Weight on the Payload-to-Fuel Flow Rate Ratio of the UH-1B, OV-10A, and X-22A Aircraft With Maximum Payload - Sea Level, Level Flight, Standard Day, Normal Rated Power.

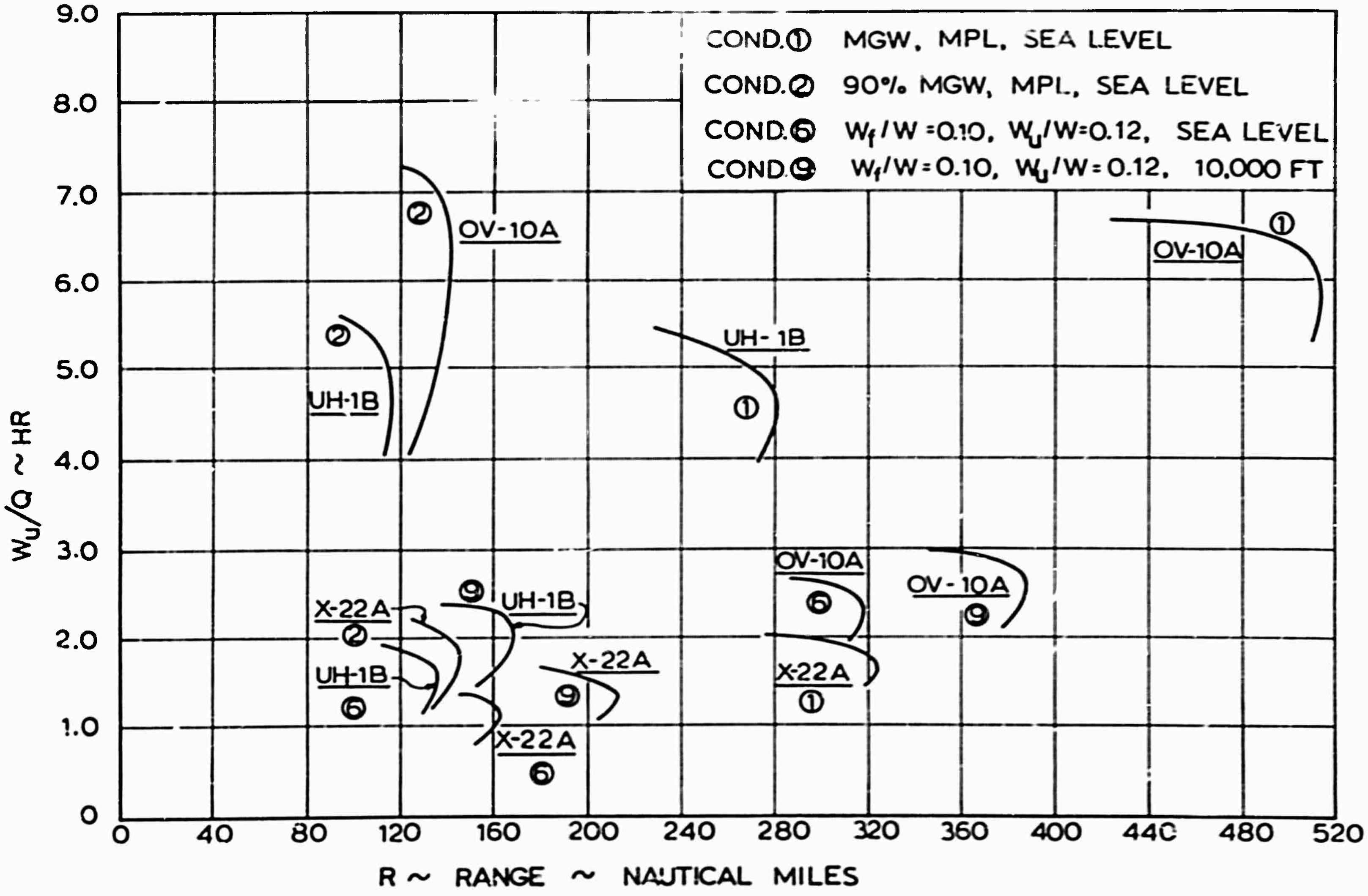

Figure 48. Relationship Between Range and Payload-to-Fuel Flow Rate Ratio for Various Payload and Fuel Load Conditions at Sea Level and at an Altitude of 10,000 Feet - Level Flight, Standard Day, Normal Rated Power.

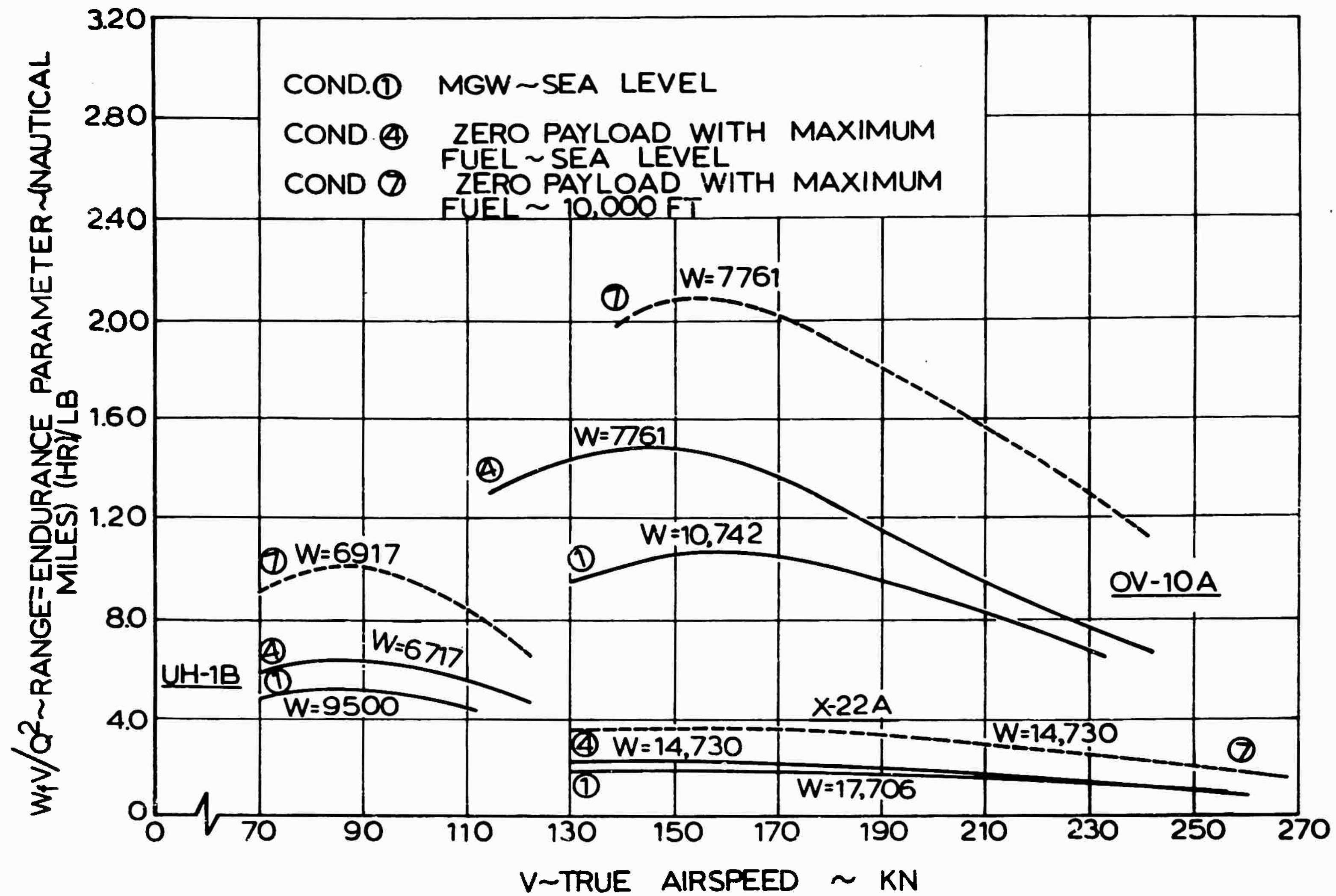

Figure 49. The Effect of Altitude and Airspeed on Range - Endurance Productivity for Maximum Gross Weight and Maximum Fuel - Zero Payload Conditions - Level Flight, Standard Day, Normal Rated Power.

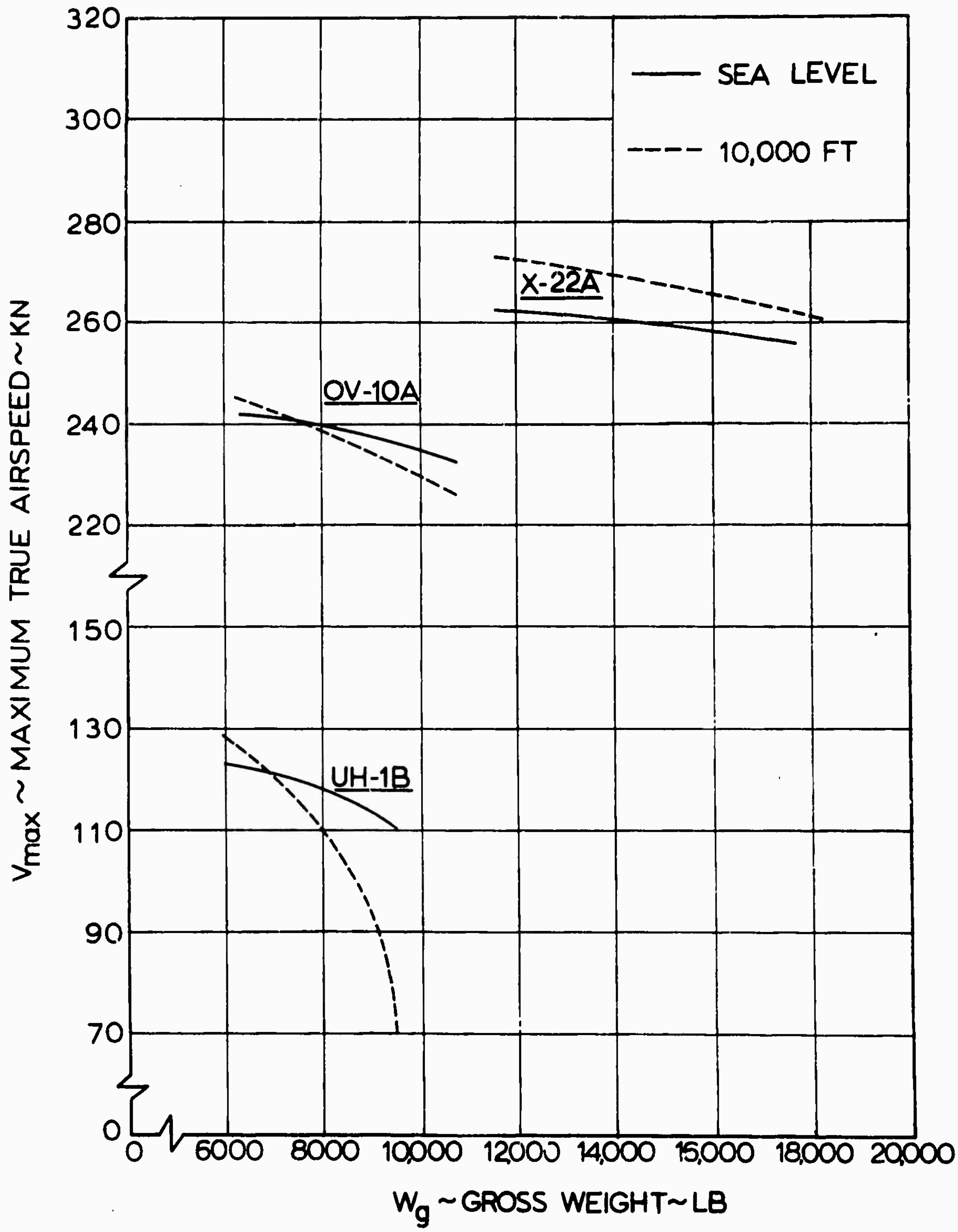

Figure 50. Effect of Altitude and Gross Weight on the Maximum True Airspeed of the UH-1B, OV-10A, and X-22A Aircraft in Level Flight - Standard Day, Normal Rated Power.

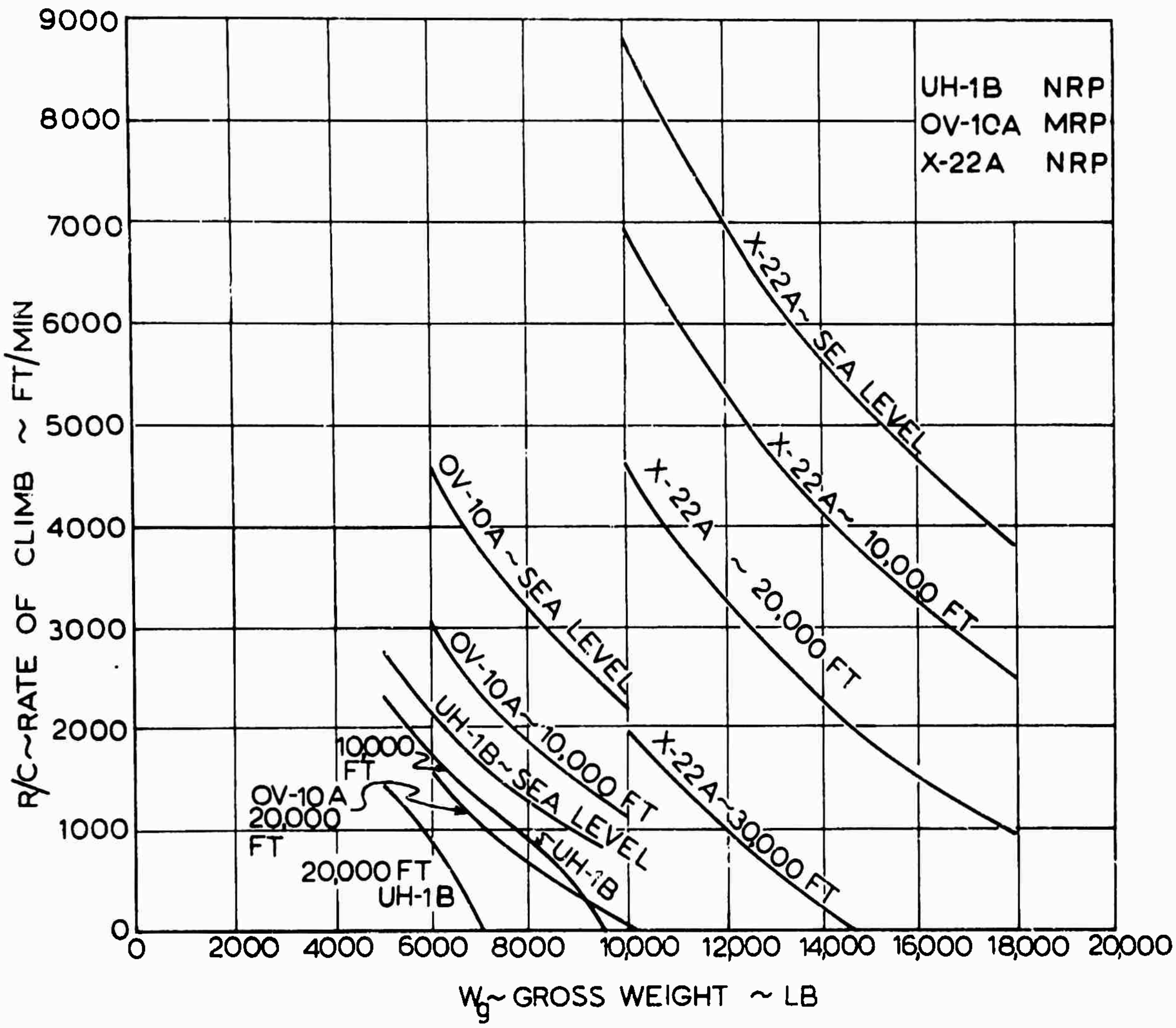

Figure 51. Comparison of Climb Rates of the UH-1B, OV-10A, and X-22A Aircraft at Various Altitudes and Gross Weights - Standard Day.

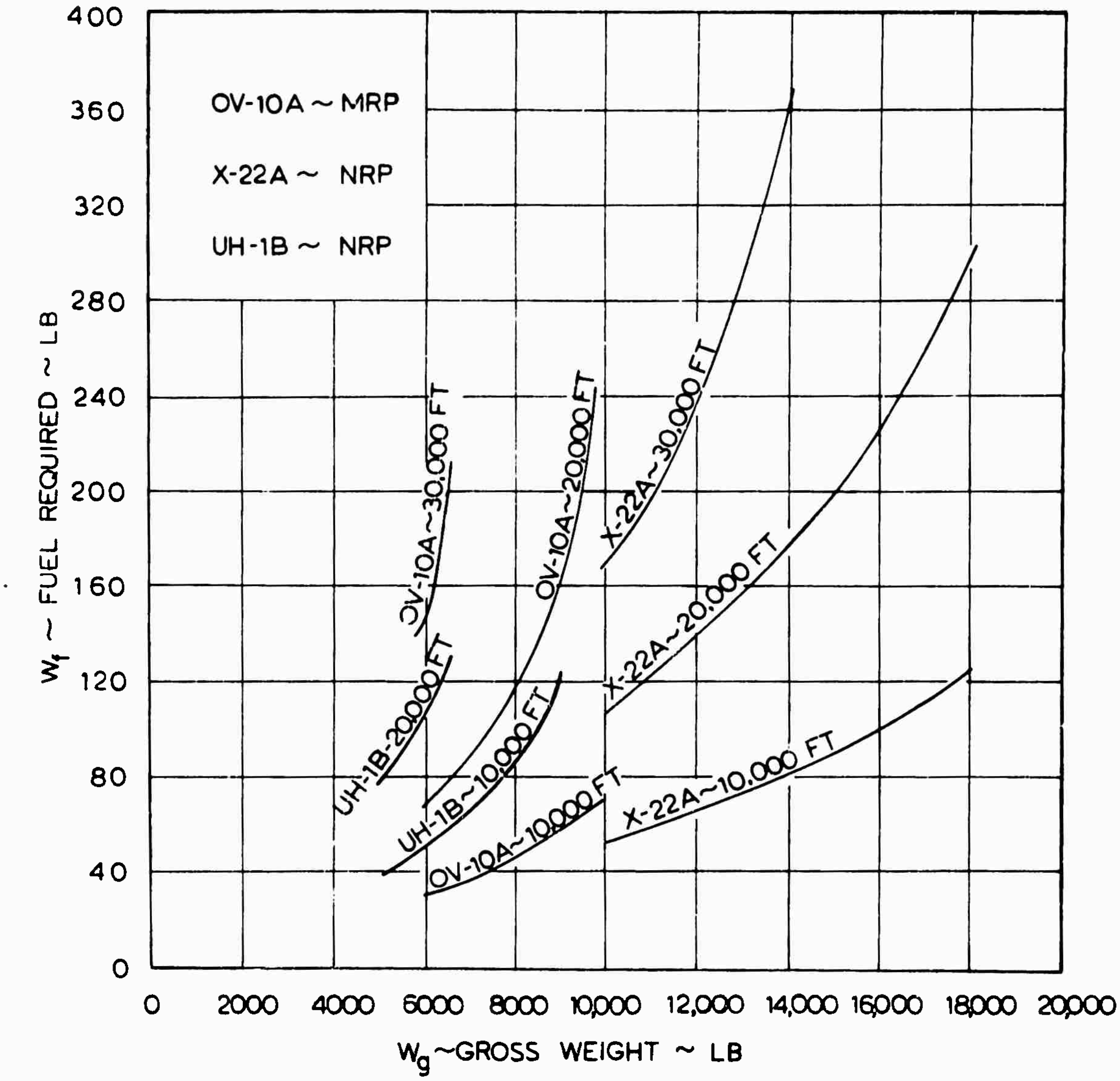

Figure 52. Fuel Required to Climb by the UH-1B, OV-10A, and X-22A Aircraft as a Function of Gross Weight and Altitude - Standard Day.

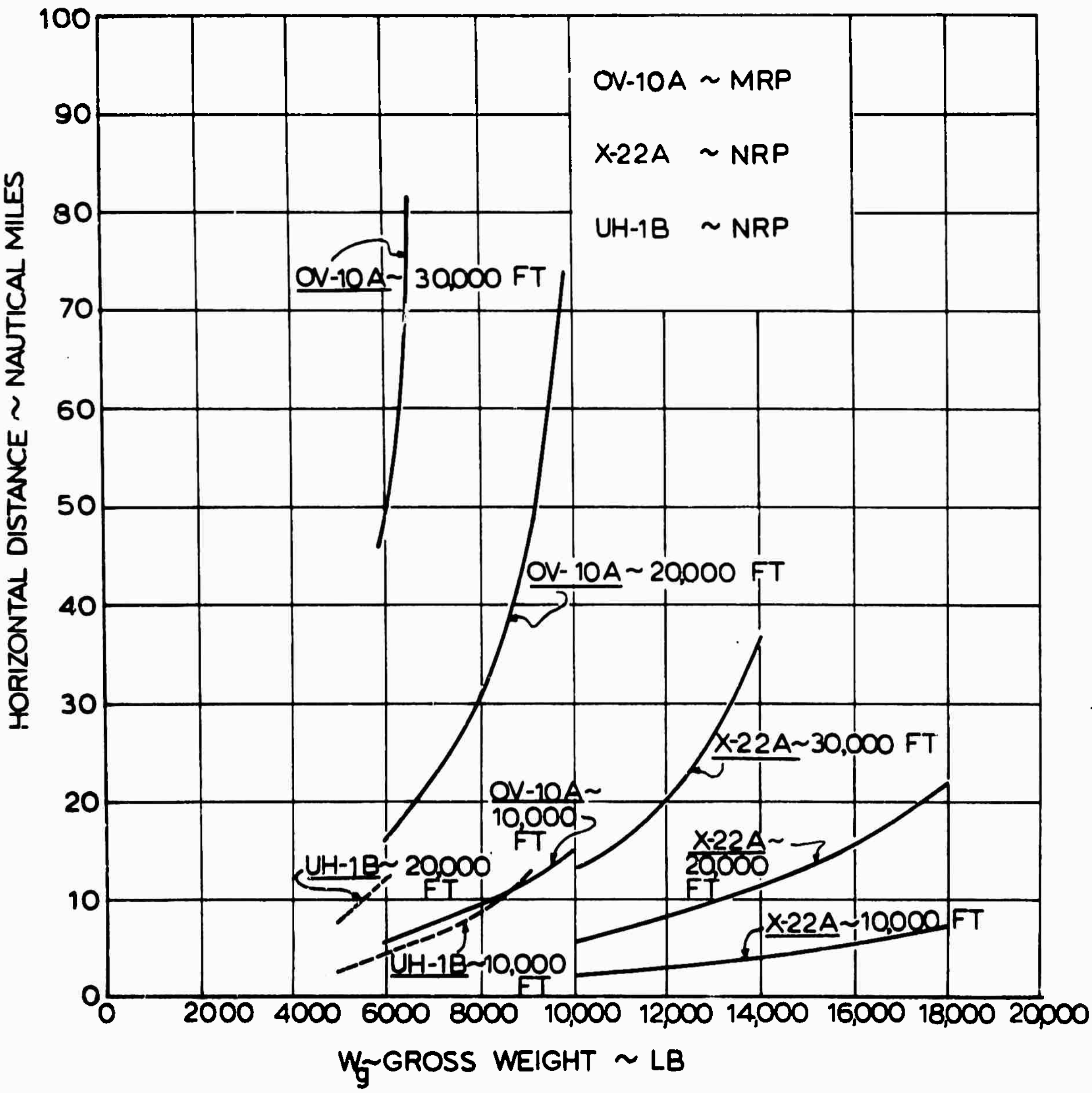

Figure 53. Effect of Gross Weight on the Horizontal Distance Traveled During Climb to Various Altitudes-Standard Day.

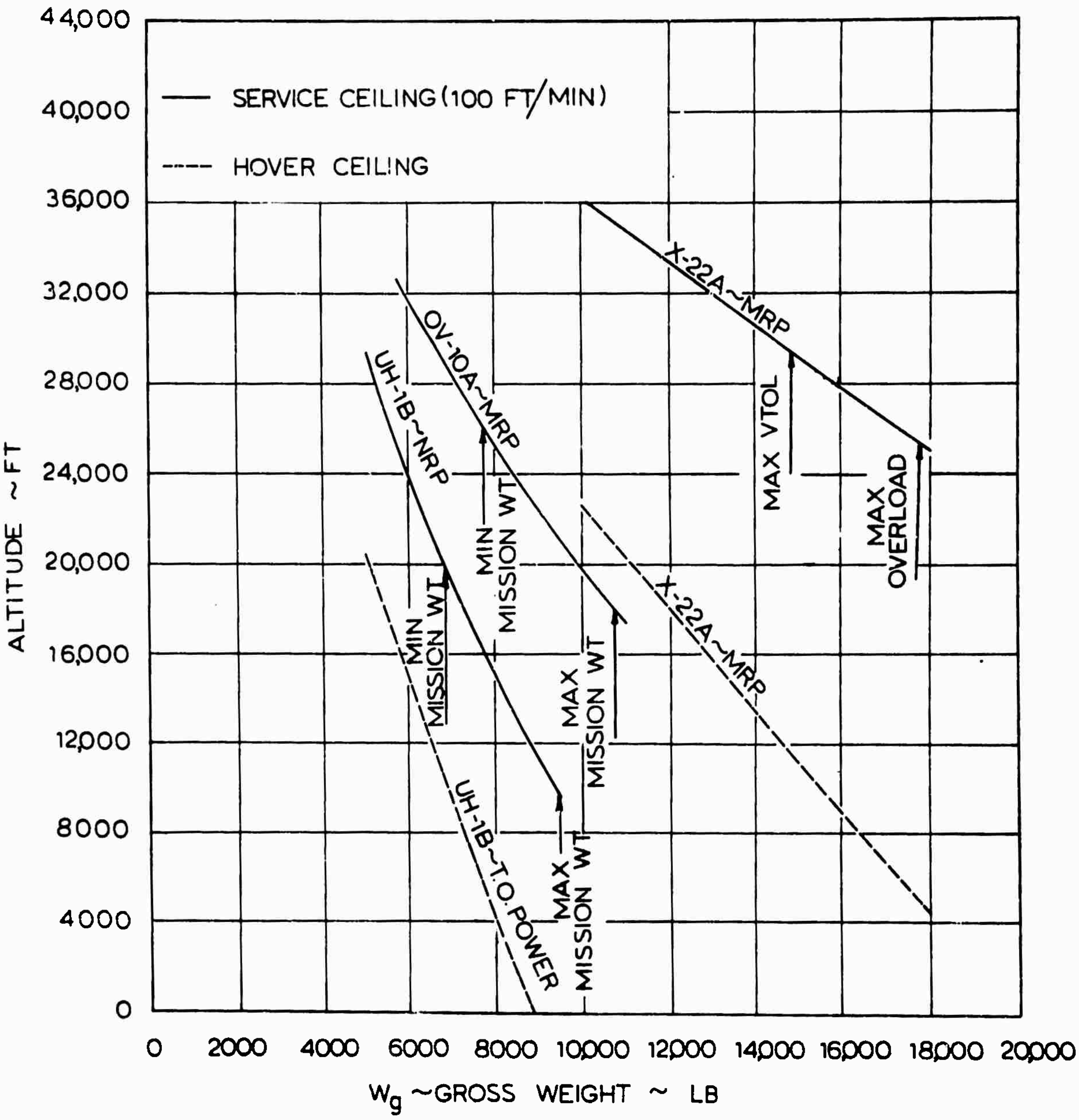

Figure 54. Effect of Gross Weight on the Service and Hover Ceilings of the UH-1B, OV-10A, and X-22A Aircraft on a Standard Day.

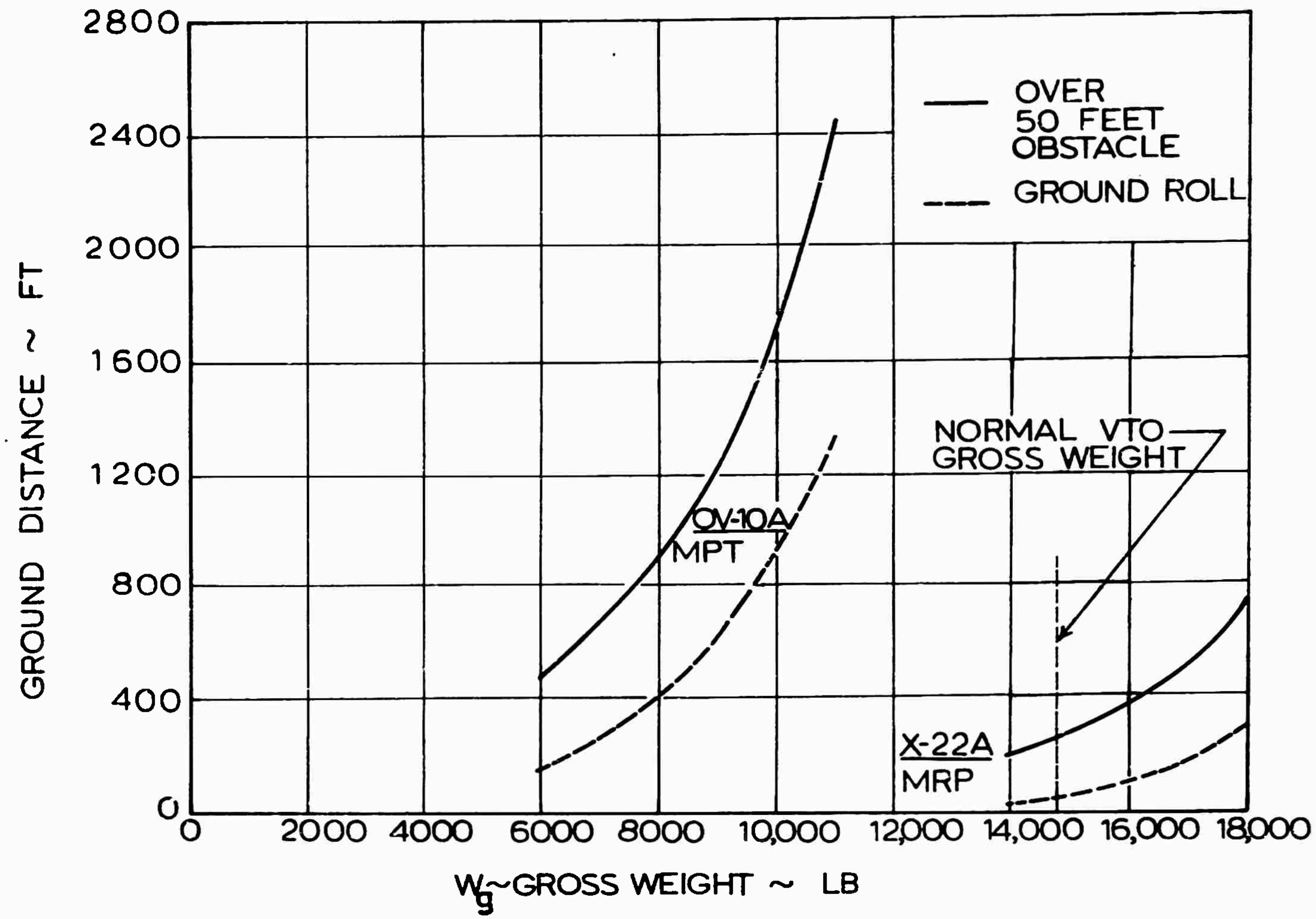

Figure 55. Effect of Gross Weight on the Horizontal Distance Traveled by OV-10A and X-22A Aircraft During Takeoff - Standard Day, Sea Level.

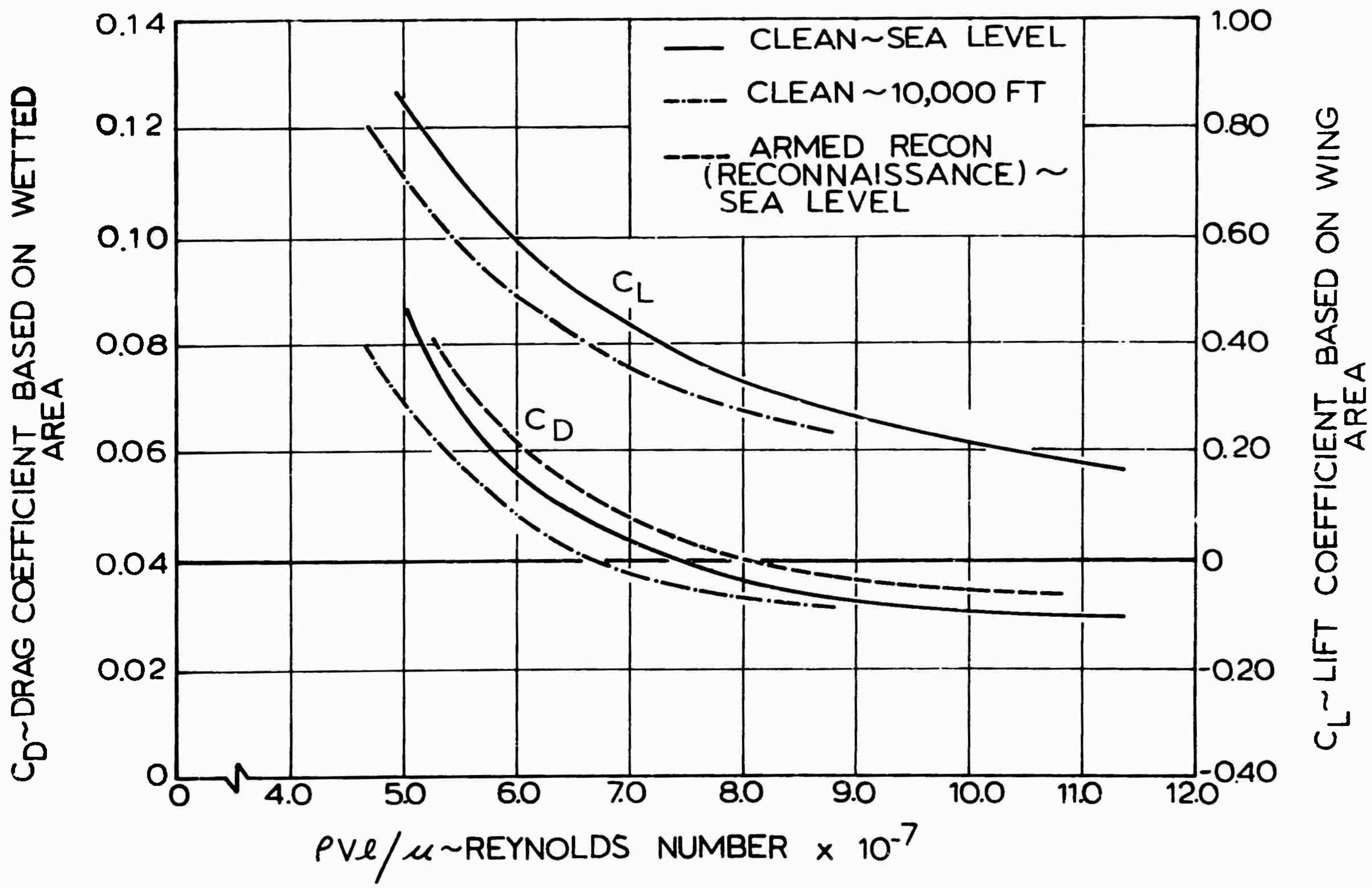

Figure 56. Variation of the Lift and Drag Coefficient of the OV-10A Aircraft With Reynolds Number - W_g = 8000 Pounds, Standard Day.

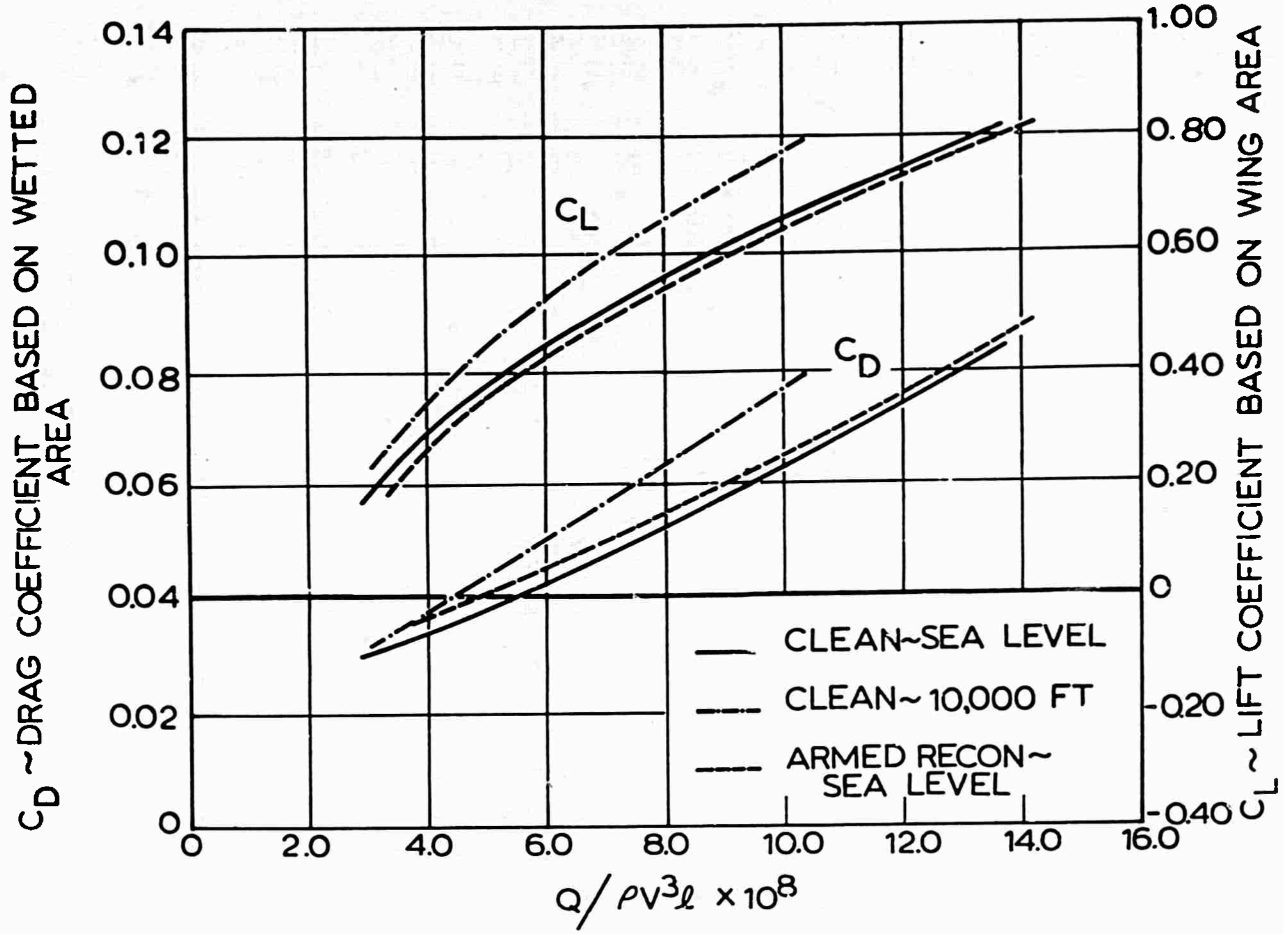

Figure 57. Lift and Drag Coefficient of the OV-10A Aircraft as a Function of Toms Number - W_g = 8000 Pounds, Standard Day.

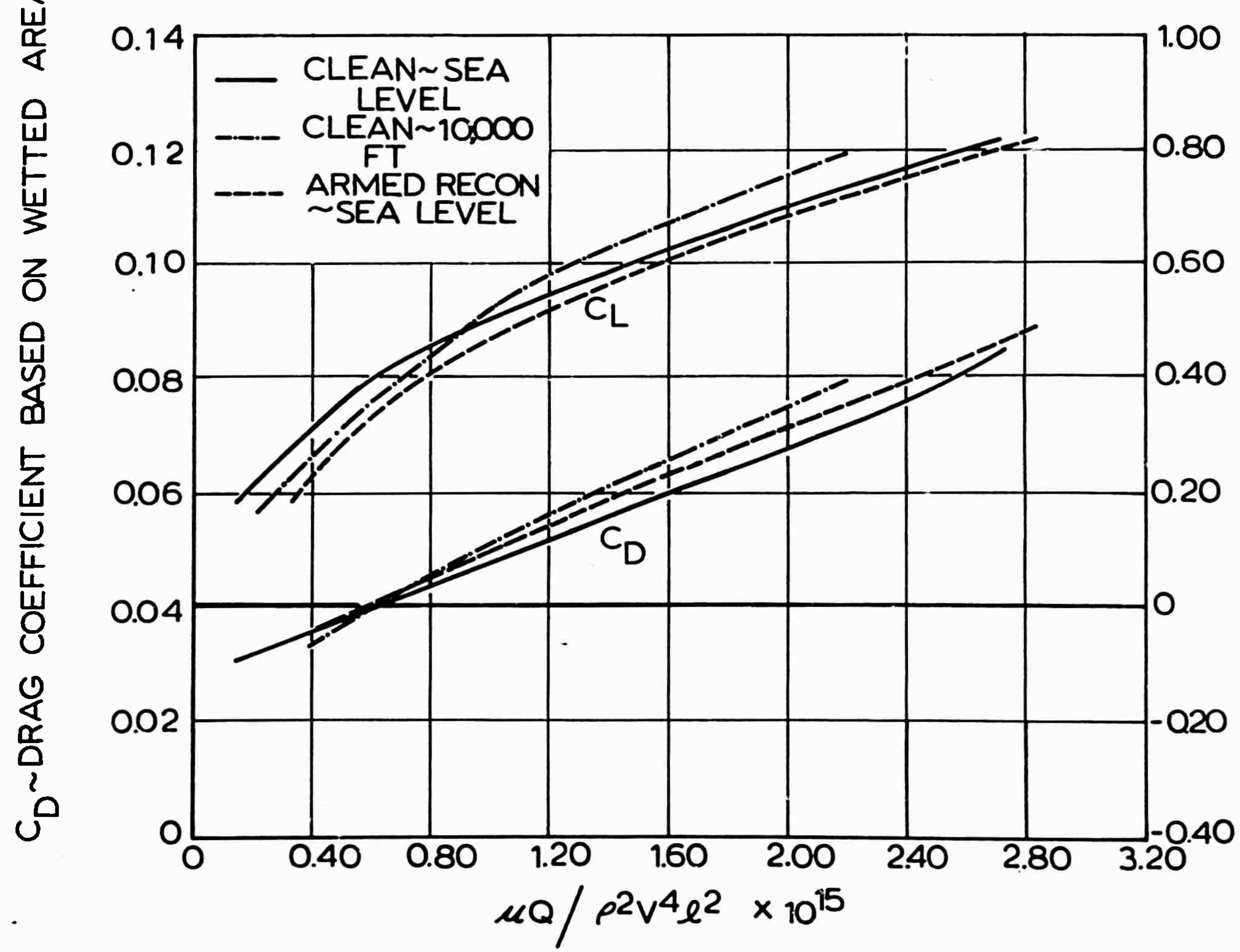

Figure 58. Lift and Drag Coefficient of the OV-10A Aircraft as a Function of a Dimensionless Number Having Fuel Flow Rate as a Variable - W_g = 8000 Pounds, Standard Day.

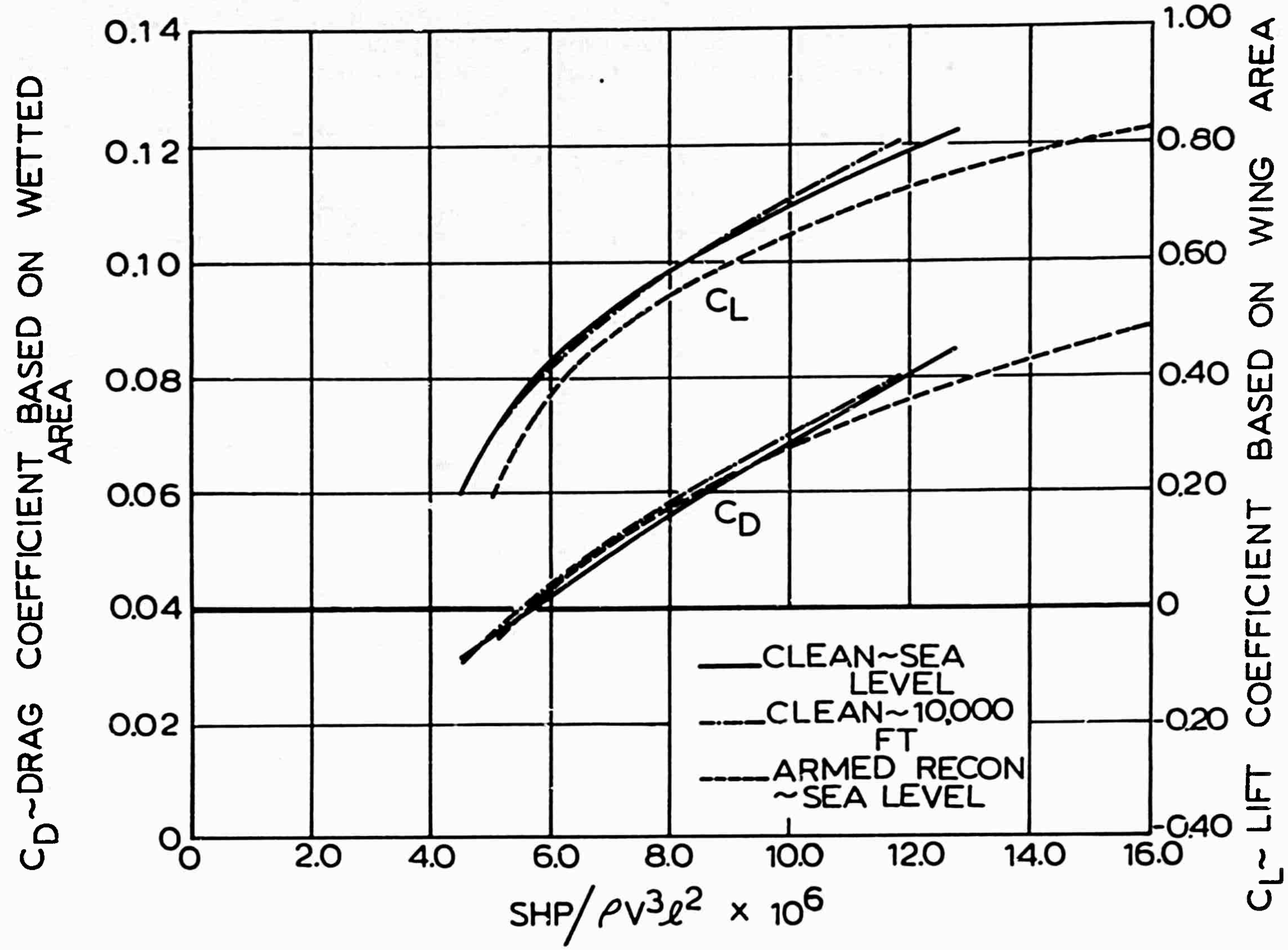

Figure 59. Lift and Drag Coefficient of the OV-10A Aircraft as a Function of Shaft Horsepower Required - W_g = 8000 Pounds, Standard Day.

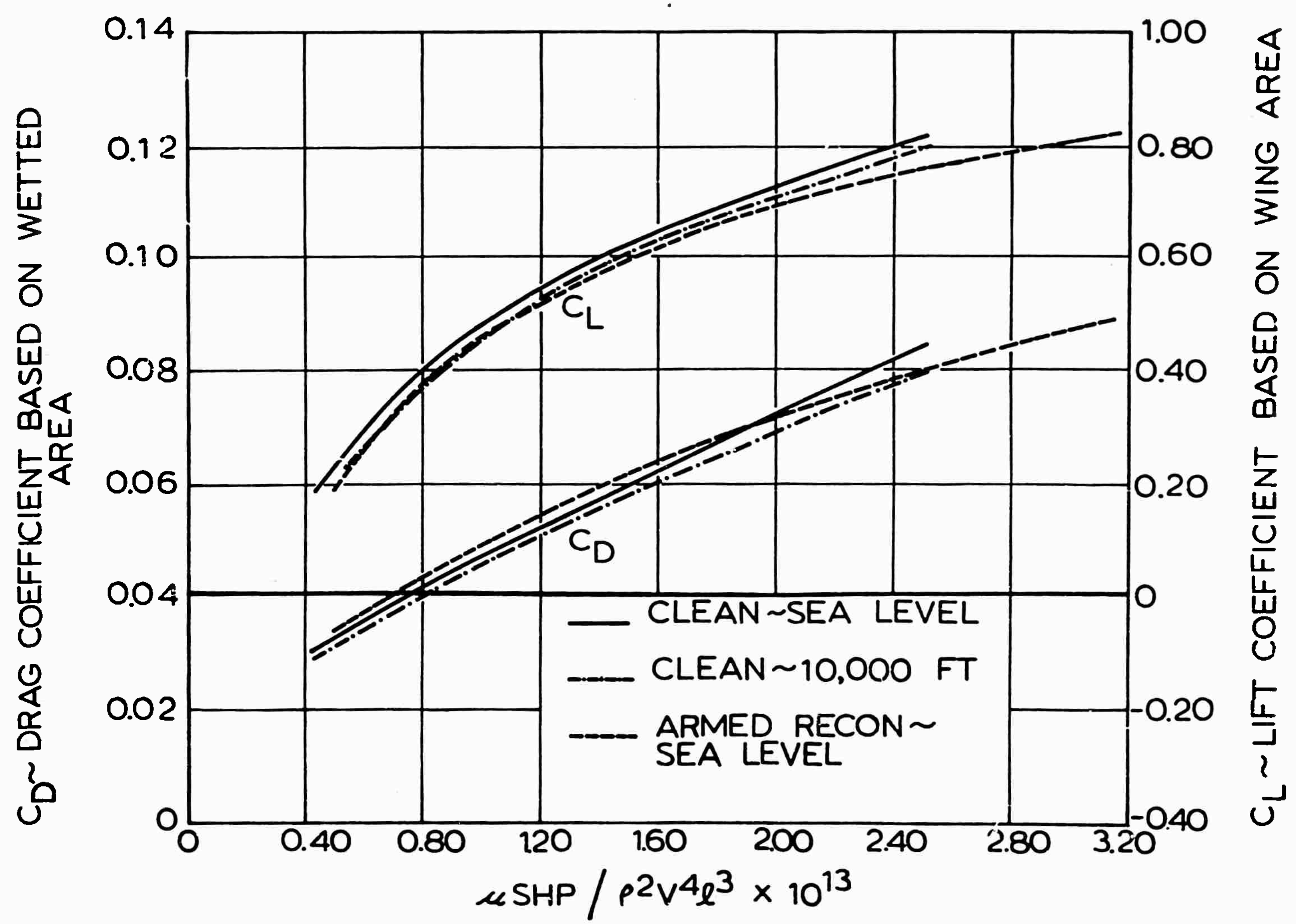

Figure 60. Lift and Drag Coefficient of the OV-10A Aircraft as a Function of a Dimensionless Number Based on Shaft Horsepower Required - W_g = 8000 Pounds, Standard Day.

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Unclassified
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Mississippi State University State College, Mississippi	2a. REPORT SECURITY CLASSIFICATION Unclassified 2b. GROUP

3. REPORT TITLE

A STUDY OF AERODYNAMIC PERFORMANCE EVALUATION AND COMPARISON TECHNIQUES FOR V/STOL AIRCRAFT

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

5. AUTHOR(S) (First name, middle initial, last name)

Donald W. Boatwright

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
July 1969	151	63

8a. CONTRACT OR GRANT NO. DA 44-177-AMC-376(T) b. PROJECT NO. Task 1F162204A14234 c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) USAAVLABS Technical Report 69-14 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) Aerophysics and Aerospace Research Report No. 77

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY US Army Aviation Materiel Laboratories Fort Eustis, Virginia

13. ABSTRACT

This report presents the results of an investigation of performance evaluation and comparison methods as applied to V/STOL aircraft. Attention is given to a number of aircraft having different sizes, gross weights, geometric configurations, and propulsion systems. Particular regard is given to the use of thermal fuel energy as a common basis for evaluation and comparison of V/STOL performance. The performance capabilities of typical V/STOL aircraft are presented and compared using both dimensional and nondimensional parameters containing fuel flow rate as a variable. In addition, three aircraft of different configurations are analyzed with regard to the effects of altitude, gross weight, and payload-to-fuel load ratio on the performance capability of each aircraft as indicated by both new and conventionally used methods. The total energy concept is discussed with regard to the optimization of climb schedules, and consideration is given to the limitations which apply to the use of nondimensional parameters which are used to describe the flow regimes of V/STOL aircraft.

DD FORM 1 NOV 65 1473 REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

Unclassified
Security Classification

Unclassified

Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
V/STOL Aircraft Comparison Techniques Aerodynamic Performance V/STOL Performance Comparison						

Unclassified

Security Classification